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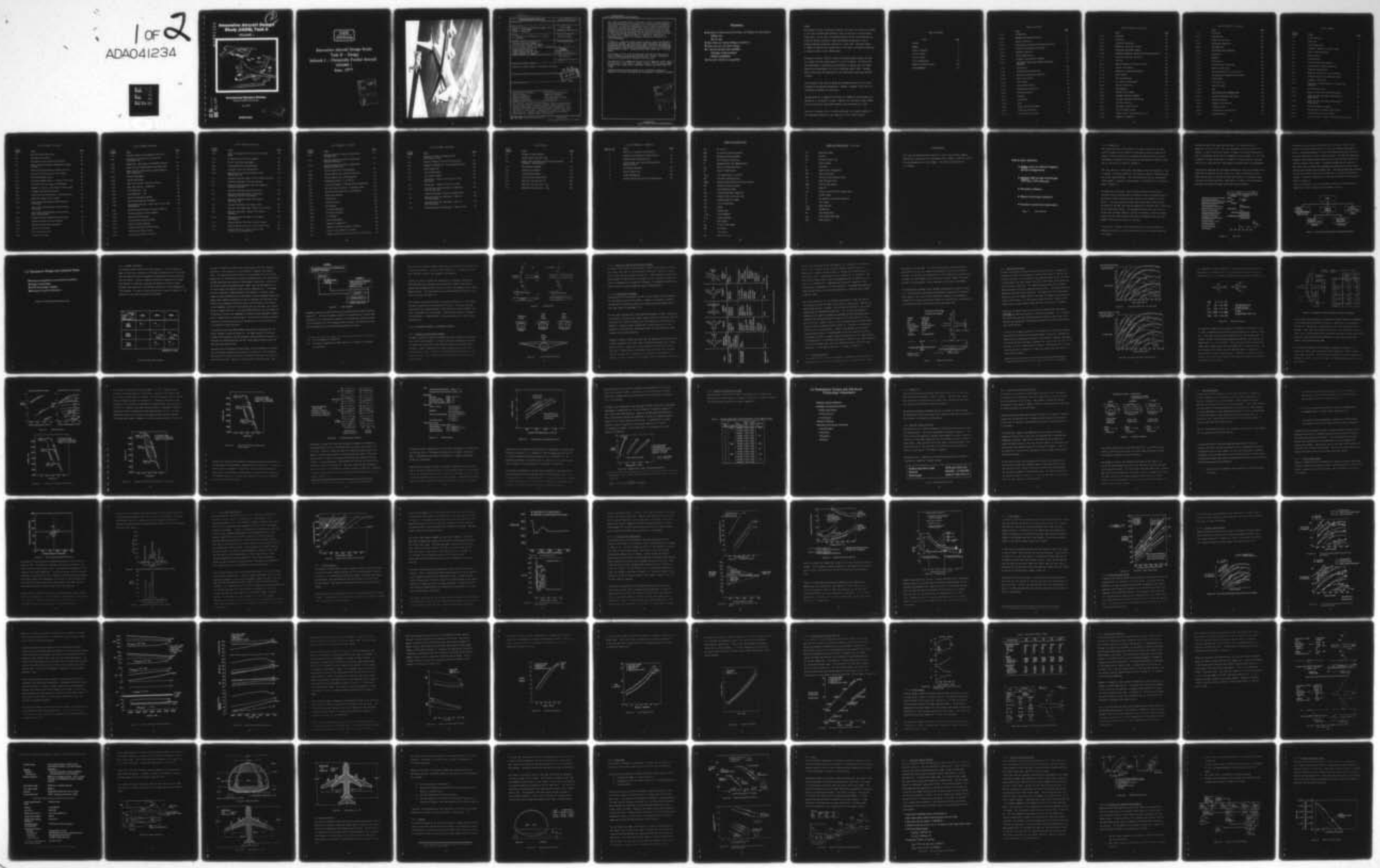

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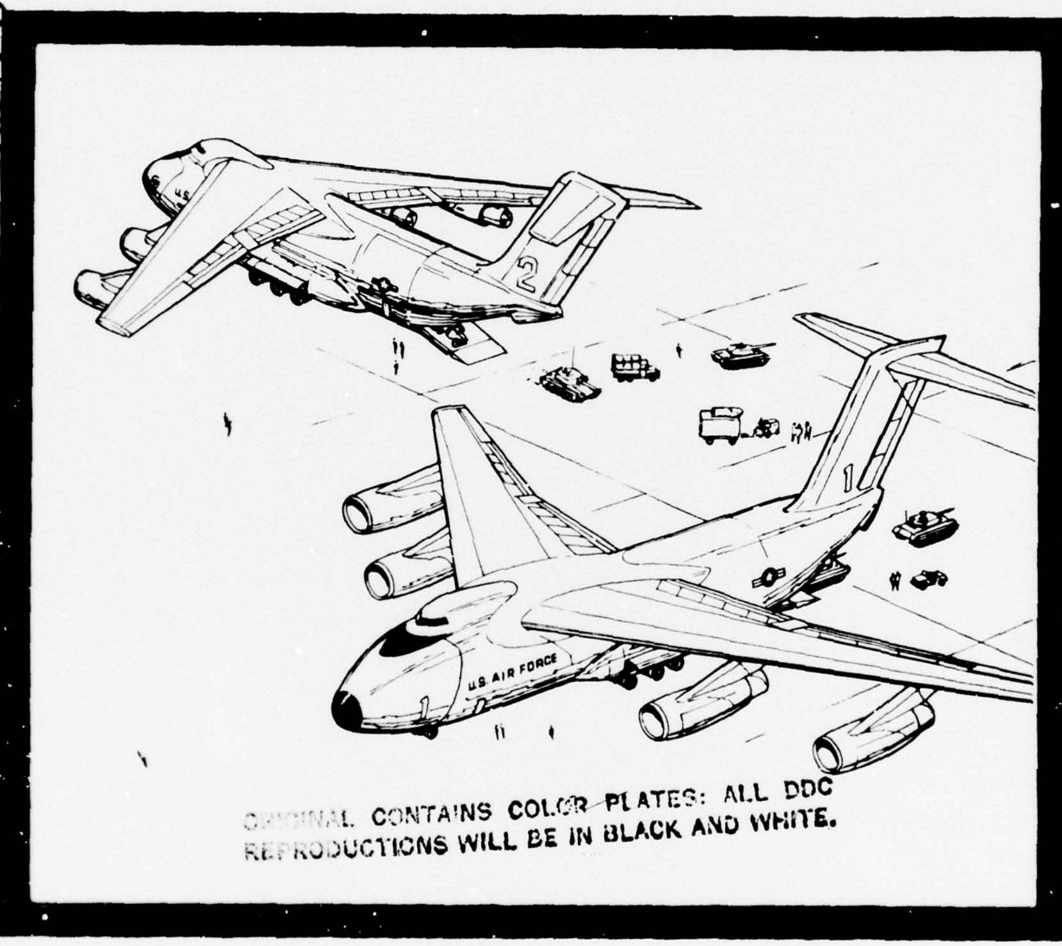
Innovative Aircraft Design Study (IADS), Task II

VOLUME I



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Innovative Aircraft Design Study

Task II – Design

Subtask I – Chemically Fueled Aircraft

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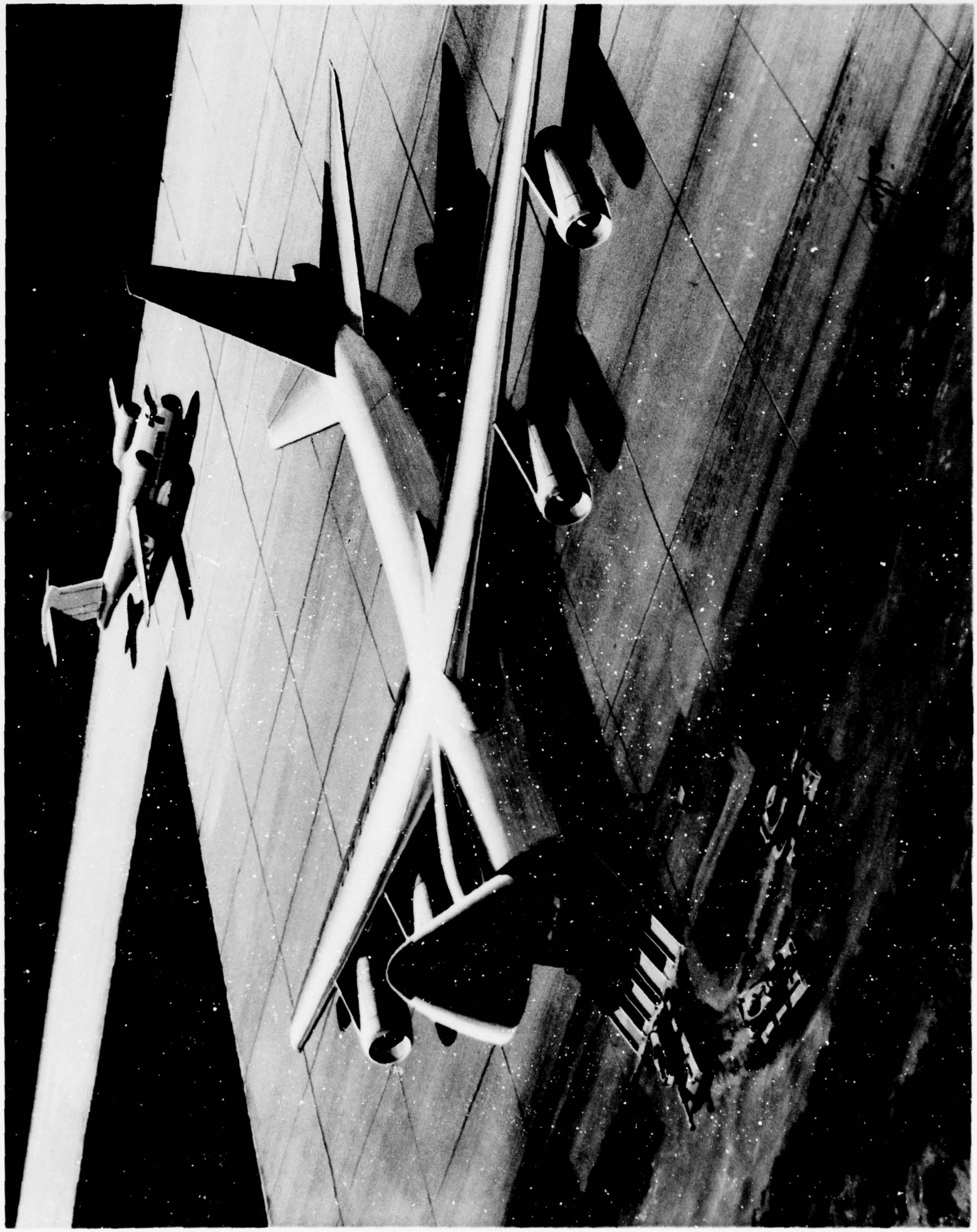

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The study was concerned with the conceptual design and evaluation of military heavy logistics transport aircraft entering service in the 1990-2000 time period. Design payloads of 200,000-600,000 lb. and design ranges of 3600-7200 nm were considered. Takeoff field length was 8000 ft. in most cases. Suitability for commercial usage was a major objective. Computer aided design techniques were employed extensively for airplane synthesis and analysis.

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The study was accomplished in two phases. Phase I included parametric design and and analysis of transports in the payload/range categories cited above. An advanced technology review, including evaluations and sensitivity analyses, was accomplished. These studies indicated that substantial gains were possible in reducing operating costs by incorporating both the low risk technology and innovative designs available in 1985, and advanced technology such as composite structures available at a later time. Additional effort is required to identify this increased cost of higher risk advanced technology to determine its cost effectiveness.

In Phase II, a baseline mission requiring 3600 nm radius and 400,000 lb. payload was selected. No difficulties were encountered in developing a configuration which met these requirements, utilizing 1985 technology. Fuselage design, wing design including aeroelasticity, and landing gear were examined in detail. Airplane takeoff gross weight was 1.5 million pounds.

The influence of fuel costs on design was examined and found to have little influence on the optimum configuration. However, increases in fuel cost did significantly increase life cycle cost.

The application of a commercial derivative to a commercial market seems to be attractive, if the market is strong. However, the long design ranges imposed by this study mission requirement imposes significant penalties in DOC.

Alternate military missions appear to be a reasonable extension of capability for a basic airplane design configured as a logistics transport.

Summary

- Benefits of advanced technology and design are substantial
  - High risk
  - Low risk
- Size does not impact design complexity
- Fuel costs do not drive design
- Commercial derivative feasible
  - Design range excessive
  - Size compatible
- Alternate missions compatible

SUMMARY

The parametric design, technology evaluation, and sensitivity analysis performed in the study indicated that substantial gains can be made in reducing operational costs, such as fuel, by incorporating both the low risk technology and innovative designs available in 1985, and advanced technologies, such as advanced composite structures available at a later time. Additional effort is needed to identify the increased cost of the higher risk advanced technology to better evaluate its cost effectiveness.

The baseline mission, 3,600 nmi radius with 400,000 pounds payload, resulted in a design which had a gross weight of 1.5 million pounds. No difficulties were experienced in establishing a configuration or design which could meet these mission requirements, utilizing a technology base of 1985. Fuselage, design, wing design and aeroelasticity and landing gear design were examined in detail.

The influence of fuel costs on design was examined and found to have little influence on the optimum configuration. However, increases in fuel cost did significantly increase life cycle cost.

The application of a commercial derivative to a commercial market seems to be attractive, if the market is strong. However, the long design ranges imposed by this study mission requirement imposes significant penalties in DOC.

The use in alternate roles of a design configured as a transport appears to be a reasonable extension of the capability of the transport design.

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SYMBOLS AND ABBREVIATIONS

A/A	Air to Air
ACLS	Air Cushion Launching System
AGE	Aerospace Ground Equipment
ATA	Air Transport Association
C^3	Command Control and Communications
C_{D_0}	Zero Lift Drag Coefficient
C_L	Wing Lift Coefficient
$C_{L_{L/O}}$	Lift Coefficient at Lift Off
DOC	ATA Direct Operating Cost
DSARC	Defense Systems Acquisition Review Council
ECS	Electronic Control System
δF	Flap Deflection Angle
IOC	Initial Operational Capability
LCC	System 20 Year Life Cycle Cost
LCN	Landing Capability Number
L/D	Lift to Drag Ratio
LBS	Pounds
LE	Leading Edge
$L\ CH_4$	Liquid Methane
$L\ H_2$	Liquid Hydrogen
M	Mach Number
M_{cr}	Critical Mach Number
MM	Millimeter
n	Load Factor
nmi	Nautical Miles

SYMBOLS AND ABBREVIATIONS (Continued)

O.W.	Operating Weight
PL	Payload
PSI	Pound per Square Inch
R	Mission Range
S	Wing Area
SFC	Specific Fuel Consumption
SHP	Shaft Horse Power
SLS	Sea Level Static Engine Thrust
t/c	Wing Thickness Ratio
TOGW	Takeoff Gross Weight
T.O.	Take off
T/W	Thrust to Take Off Gross Weight Ratio
V_f	Flutter Speed
V_H	Max Speed for Continuous Operation
V_L	Limit Speed
V_{rqd}	Speed Required
A wet	Wetted Area
AR	Wing Aspect Ratio
Λ_{LE}	Wing Leading Edge Sweep
W/S	Wing Loading

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Task II study objectives

- Define most cost effective logistics aircraft configurations
- Identify high leverage technologies 1985 base; 1995 advanced
- Secondary missions
- Mission/technology sensitivity
- Examine commercial commonality

Figure 1-1. Study Objectives

1.0.0 INTRODUCTION

The Innovative Aircraft Design Study is an effort sponsored by the Aeronautical Systems Division, Deputy for Development Planning (ASDXR) United States Air Force, which has as its purpose the identification of mission requirements, conceptual aircraft design and technological innovations leading to a new family of logistics transports.

The study consists of three tasks; requirement, design and evaluation. Task II, Design, is further divided into two subtasks; chemically powered designs and nuclear powered designs. This report describes the work accomplished under subtask I, chemically fueled designs. The objectives of the study are shown in Figure 1-1.

The objectives of the study were to identify the most cost effective logistics aircraft configurations with emphasis given to the application of the design to other military missions and also to the commercial freight market. It was also desired to assess those technologies which might be available for the 1985 time frame and beyond and to determine their contribution to the system effectiveness. The secondary missions which were to be considered were strategic offensive, tactical and command, control and communication, with the intent being to show that a procurement larger than that needed for the logistics mission is feasible.

It was also of interest to show the sensitivity of a baseline design to offdesign variations in mission characteristics such as range, payload and field length.

Considerable mention has been made, Reference 1, of the application of commercial airlift to fulfill the surge requirement for the Military Airlift Command. It was of particular interest to attempt to evaluate the attractiveness of a commercial derivative of an advanced military transport in fulfilling a commercial freight market need, as a means of increasing the buy, reducing the unit cost, and providing a surge capability through CRAF-like arrangements.

The study was organized into two phases separated by a decision milestone which identified the baseline mission, Figure 1-2. Phase I consisted of a parametric design effort to examine the broad implications of the range of study conditions which were specified. Concurrently, in Phase I, the technologies applicable to advanced transport design were being evaluated.

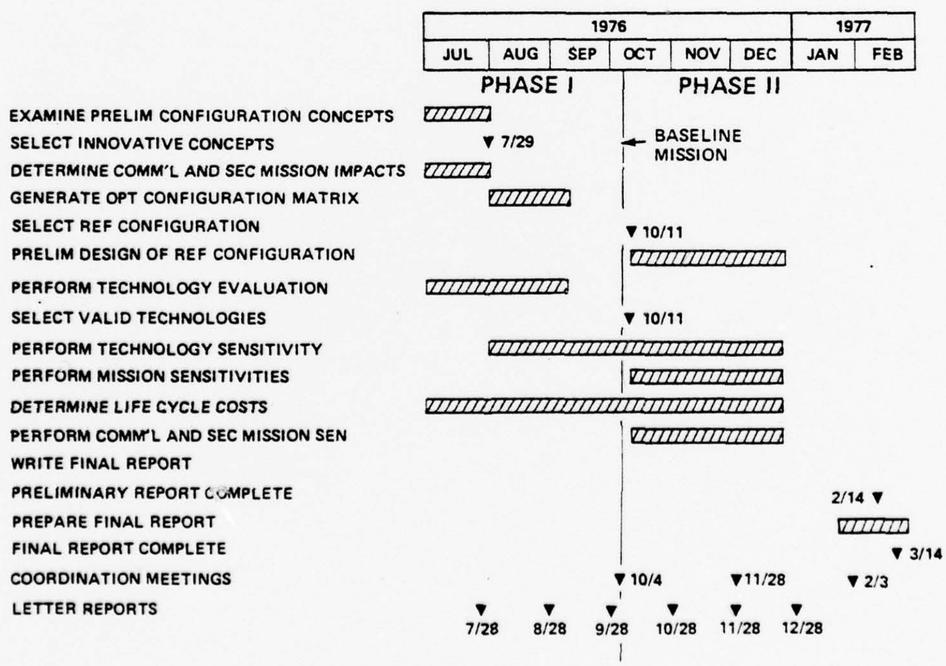


Figure 1-2. Study Plan

Following the mission selection, Phase II was initiated. Phase II was a classical Preliminary Design effort in which the information generated in Phase I, the design parameters, and the 1985 technology baseline, were used as a basis upon which to select a Baseline Design. That Baseline Design was subjected to more detailed design analysis including structural analysis of the wing, body and landing gear. The Baseline Design was also used as a basis upon which to examine the technology and mission sensitivities and to evaluate the alternate mission capabilities. The baseline design was validated and cycled to confirm the initial configuration or instigate configuration changes. The overall organization of the study consisting of three tasks is shown in Figure 1-3.

Task III, Evaluation, will be an in-depth evaluation of the capabilities identified in Task II against the requirements of Task I.

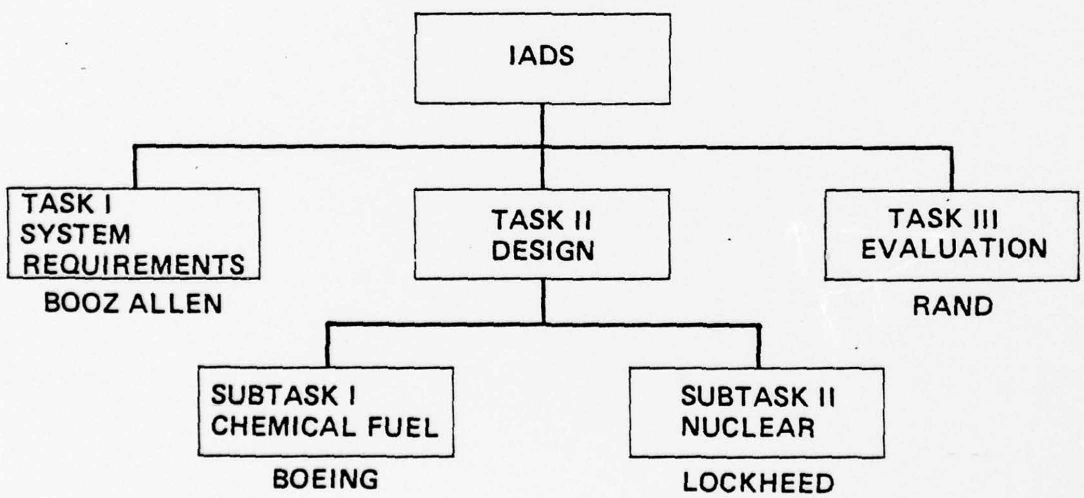


Figure 1-3. Innovative Aircraft Design Study Organizational Outline

2.0 Parametric Design and Analysis Tasks

- General arrangement and reference baseline
- Design sensitivities
- 1985 technology baseline
- Parametric matrix definition

Figure 2-1. Parametric Design and Analysis Tasks

2.1.0 GENERAL DISCUSSION

The Parametric Design and Analysis task, Figure 2-1, was instituted as a means of examining a broad basis of payloads and ranges with which to understand the impact of mission parameters on maximum take off gross weight and fuel utilization. A matrix of design conditions was desired which ranged from 200,000 lb to 600,000 lb payload, and 3600 nmi to 7200 nmi range. Alternate fuels were also to be examined/ specifically liquid hydrogen and liquid methane. The matrix of design conditions is shown in Figure 2-2. The design rules were those specified in MIL-C-5011A.

Payload (lbs) / Range (nmi) (field length)	200K	400K	600K
3600, (8000)	JP	JP	
5500, (8000)		JP / LH_2 / LCH_4	JP / LH_2 / LCH_4
7200, (9000)		JP	JP

MilC5011 A rules

Figure 2-2. Phase I Study Conditions

In order to accomplish a study of such broad scope, within the resources available, it was necessary to utilize methods of computer aided design (CAD). This approach matched the need for a broad based survey of airplane designs and the interrelation of advanced technology levels with those designs. The CAD system which was applied to the parametric design was a variation of the Airplane Engine Requirements System (ARES) which was developed for the Air Force Propulsion Laboratory under contract, Reference (2). This method utilizes a statistical sampling method, Latin Squares, to reduce a large number of design permutations and combinations to one of manageable proportion. Second order regression equations are used to represent the functional relationship of the dependent variables, such as mission range or gross weight, to the independent variables of the problem such as wing sweep, wing loading, thrust to weight ratio, etc. The current ARES formulation has the capability of treating a field of ten independent variables for each dependent variable, which is identified as a figure of merit. By use of Latin Squares the design combinations and permutations can be represented by 121 design points with an acceptable and specified error.

The process by which the ARES parametric designs were integrated into the overall program is shown on Figure 2-3. The total sub-task I effort was divided into two phases, with the parametric designs providing the basis for selection of a Baseline Mission and the initial design characteristics for the Preliminary Design Phase.

Concurrent with the Parametric Design Phase, an effort devoted to an assessment of advanced technology was carried out to establish a 1985 Technology Baseline and to provide an evaluation of advanced technology which might be available after 1985. Because of the concurrent technology effort, some

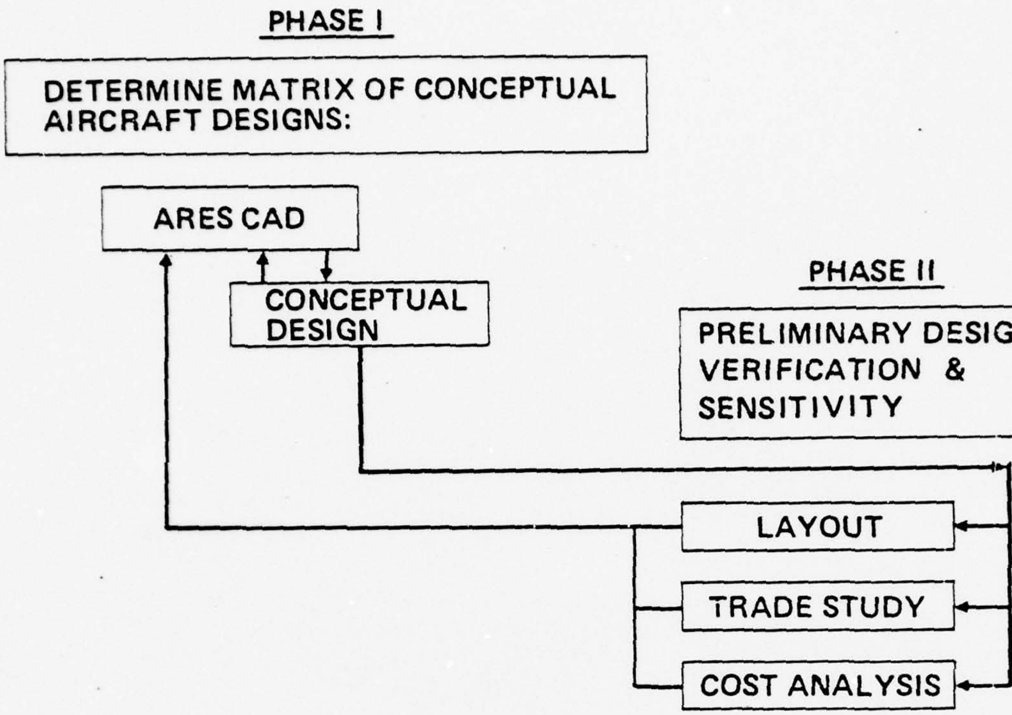


Figure 2-3. Study Approach

parametric analysis was carried out prior to establishing the 1985 Technology Baseline[1]. This set of parametric data[2] was generated for a current technology basis with the exception however, that the propulsion system was based on the availability of a new advanced turbo fan engine, and provided the basis for the generation of the 1985 technology parametric designs.

---

(1) This corresponds to a 1990 IOC.

(2) This data was generated on IR&D funds and is included in this report for continuity.

upon which to extrapolate weights, aerodynamics, and propulsion characteristics to large gross weights. Figure 2-4 shows schematically the general arrangements selected for the JP and cryogenic configurations.

A particular area of interest was the analysis of fuselage cross-section. Structural penalties associated with the design of durable structures for pressurized fuselages dictates the use of circular arc segments to the largest degree possible. The cross-sections which were examined were double arc and circular as shown on Figure 2-5.

The cryogenic configuration differs from the JP configuration in two respects: the body cross-section and the bracing of the wing. The double lobe body cross-section was selected because of the ease with which the cryogenic fuel tank is integrated with the fuselage. Since the wing carries no cryogenic fuel and obtains no bending relief, a strut braced wing is a logical design characteristic.

2.2.0 CONFIGURATION CONCEPT AND REFERENCE BASELINE

2.2.1 Innovative Configuration Concepts

A number of possible innovative configurations were reviewed for application to military cargo transports. In order to provide a broad based assessment of innovative configurations, the following configuration concepts were reviewed: canard, tandem wing, tail-less, oblique-wing, ram wing and air cushioned landing gear. Additional configuration features discussed in the main part of this study include strut braced wings and bodies with various cross sections.

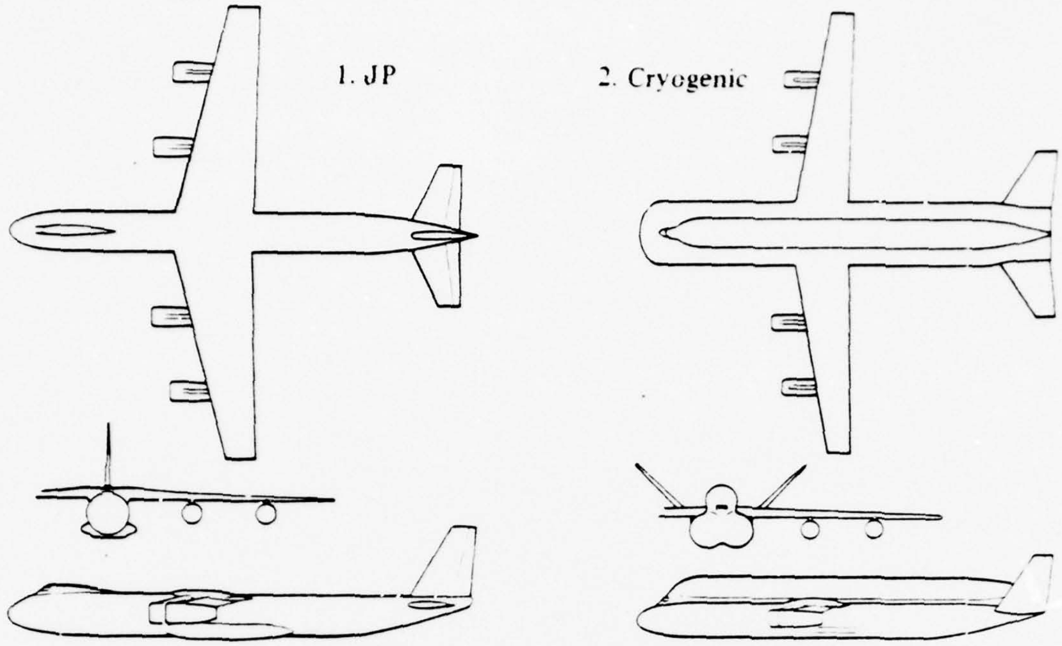


Figure 2-4 Configurations

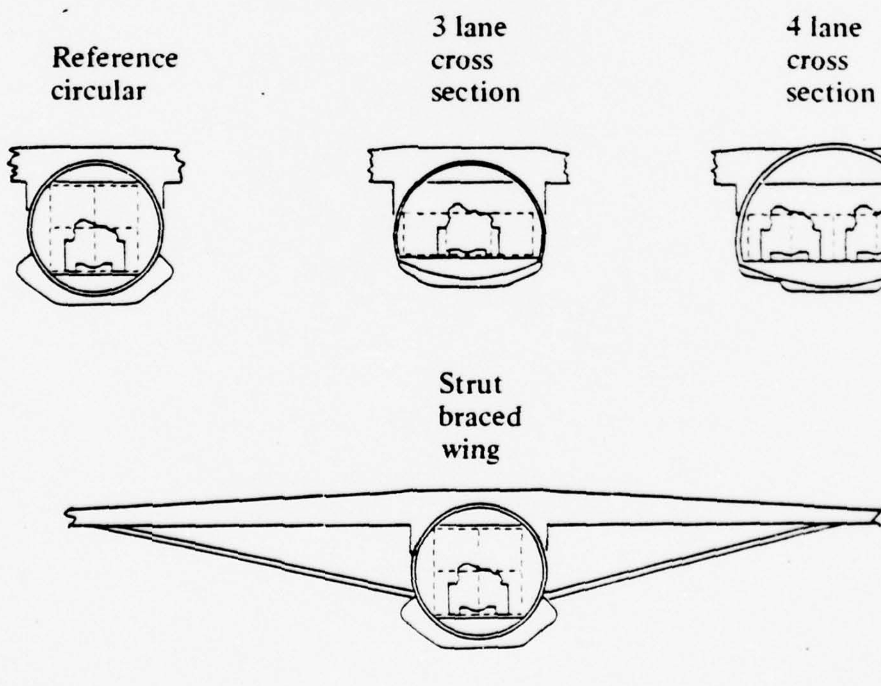


Figure 2-5 Study Fuselage Cross-sections

2.2.1.1 Results of Reviews of Innovative Concepts

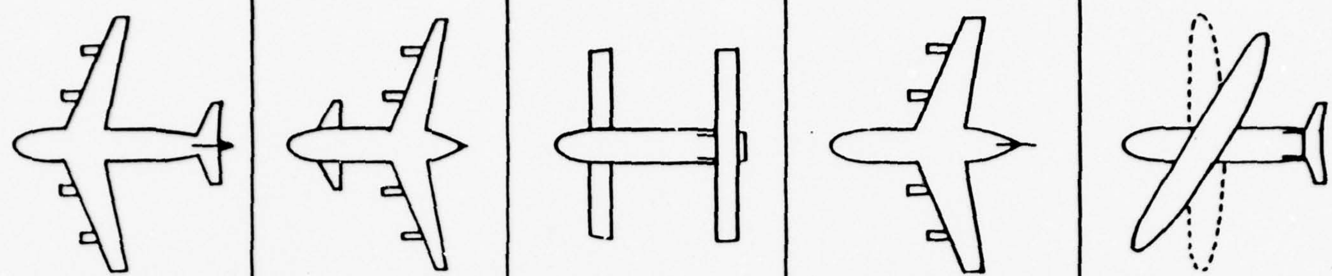
Estimates of the two main figures-of-merit, takeoff gross weight and minimum fuel burned, were made for the candidate configurations (except for the ram wing) and the results are shown on Figure 2-6. These candidate configurations were reviewed for applicability to strategic airlift, alternate military missions, and adaptability to commercial freighters. No predominant benefit was found to arise from any of these more unusual configurations.

2.2.1.2 Discussion of Concepts

Configurations utilizing canards and the tail-less concept have the potential for improvement relative to the others. However, all innovative concepts reviewed were considered unsuitable after weighing these advantages against the higher degree of development risk.

For the canard configuration, items needing development include: canard high lift devices, fail-safe flight control system, potential directional stability, pitchup and roll control problems due to vortices shed on the fin and wing. Commercial transport studies indicate that a canard may save 5% in fuel burned over the conventional configuration when the landing approach speed dictates the wing area selection.

Passenger transport studies have shown that the tandem wing configuration has a higher drag-due-to-lift factor than a comparable monoplane, and its empty weight fraction is higher due to crash protection provisions for the high rear wing. Another possible arrangement, where the front wing is mounted high and the rear wing is mounted low, may reduce this weight penalty. However, its fuel utilization will be greater than the reference configuration.



Ratio	Reference configuration	Canard	Tandem wing	Tailless**	Oblique wing
Advantages	Well understood design Good overall performance and mission flexibility	Potential for lower gross weight	Possibility of wing natural laminar flow	High cruise range factor	Mission versatility
Disadvantages and problems	None apparent	Undesirable stall characteristics Fail safe systems essential Direct'l stability problem due to vortex interference on fin Canard high lift devices required	High empty weight fraction (high mtd rear wing) Balance problem High drag-due-to-lift	C_L max 65% of conventional Loadability limitations Stability in pitch and yaw Ground clearance angle	Body mtd engines Balance Problem Wing mtd engines High interfer drag Mechanical arrangement problems Stability and control cross coupling
Development risk	Low	Medium	High	High	High

**All payload in body

Figure 2-6. Configuration Concept Comparisons

The tail-less configuration has the potential for 10 percent or more improvement in fuel efficiency but has technical questions needing resolution. Development is needed to define a means to increase the trimmed maximum lift coefficient, which is approximately 65% of conventional configurations. This configuration needs careful stability assessment as well as attention to its Dutch Roll characteristics. Depending upon the specific geometry of the configuration, the potential for spinning exists. The loadability aspects of these configurations may be restricted because of a relatively small center-of-gravity range.

The cruise range factor for the oblique wing aircraft is about the same as the reference aircraft for equal wing structural aspect ratios. Because the empty weight fraction is greater, the resulting fuel burned will be slightly greater than the reference configuration. The cruise at low supersonic speeds does not produce a performance advantage for the cargo application, therefore the high speed capability aspect of this configuration is not a positive factor. However, use of a small wing sweep angle to increase endurance would add to alternate mission versatility. Location of the engines is difficult on this configuration. Installing the engines on the aft body causes balance problems. Engines mounted on the wing require a swiveling arrangement and result in nacelle-pylon interference drag. For these reasons, this configuration is considered unsuitable for large military cargo transports. A June 1976 Lockheed study, which was sponsored by NASA, came to the same conclusion. A more detailed analysis is included in Appendix A.

2.2.2 Reference Baseline

As a result of the review of those general arrangements which might qualify for application to a logistics transport design, a conventional four engine configuration was selected. All other configurations involved additional development risk which was viewed as unacceptable relative to the benefits involved. In addition, the extrapolation of the configuration characteristics to regions of gross weights, which have not as yet been studied, could best be made on a configuration in which there was a high level of confidence.

The configuration selected as the reference configuration is shown in Figure 2-7. This particular configuration was selected on the basis of minimizing gross weight. The circular cross section was selected for the parametric studies for its ease of analysis and simplicity, realizing that the sensitivity of the design to body cross section would be evaluated later.

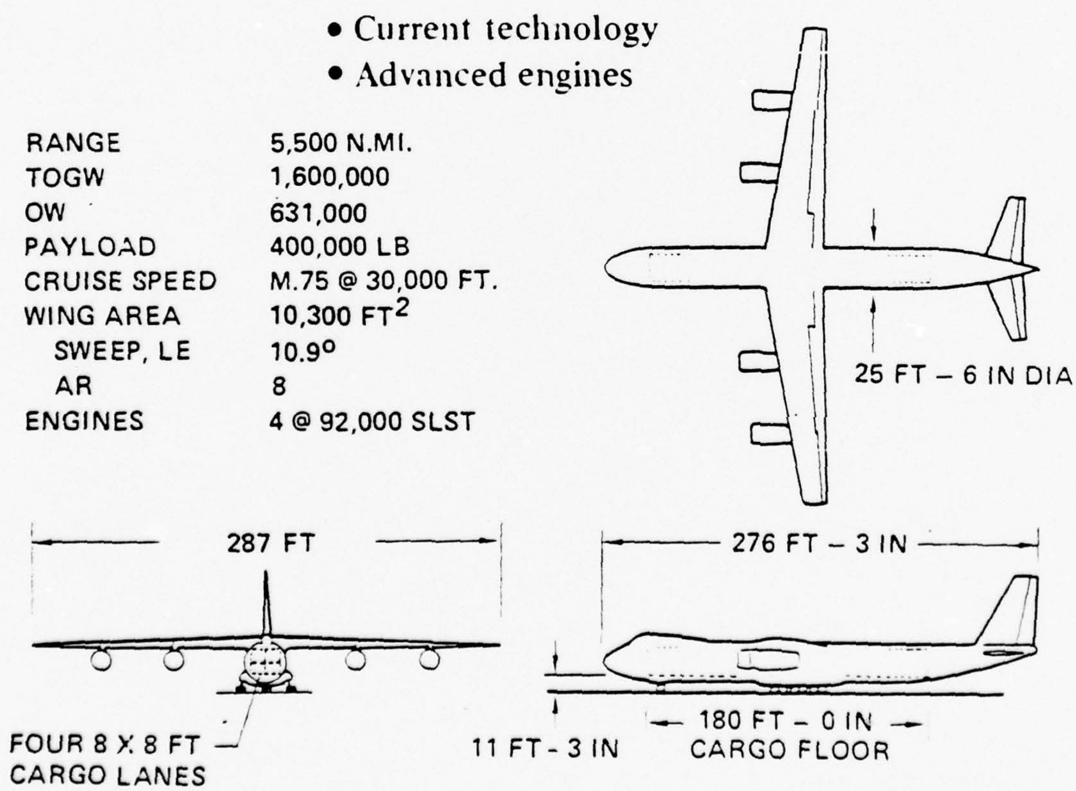


Figure 2-7. Reference Configuration

2.2.3 Design Sensitivities

Although the parametric designs were established using CAD, a reference configuration was drawn and analyzed at a nominal design point of 5500 nmi and 40,000 lbs payload. A sizing analysis was performed combining the variations of wing loading, and thrust weight ratio with field length, gross weight and fuel efficiency, Figure 2-8. This composite design chart is for planform design characteristics which maximize fuel efficiency. The design points are defined by the field length constraint. The design point for minimum gross weight occurs at the tangent point of the lines of constant gross weight and constant field length. All other points along the line of constant field length requires a gross weight higher than that of the point of tangency.

Similarly, the tangent point with the lines of constant fuel efficiency, $\left(\frac{\text{Ton-Mile}}{\text{lb. fuel}}\right)$, provides the sizing for a fuel efficient design. It is interesting to note that the fuel efficient designs invariably require lower wing loadings and consequently lower thrust-to-weight ratios than do the minimum gross weight designs.

A similar set of design solutions is illustrated for a design with planform characteristics chosen to minimize gross weight, Figure 2-9. Although the design point for minimum gross weight is not strongly influenced, the impact for the maximum fuel efficiency increases from a value of 2.2 to over 2.7 ton miles/lb. of fuel or approximately 20 percent. The design point for a fuel optimized design also occurs at a lower wing loading than does that for a range optimized design.[1]

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[1] Range optimized and minimum gross weight are taken to be synonymous.

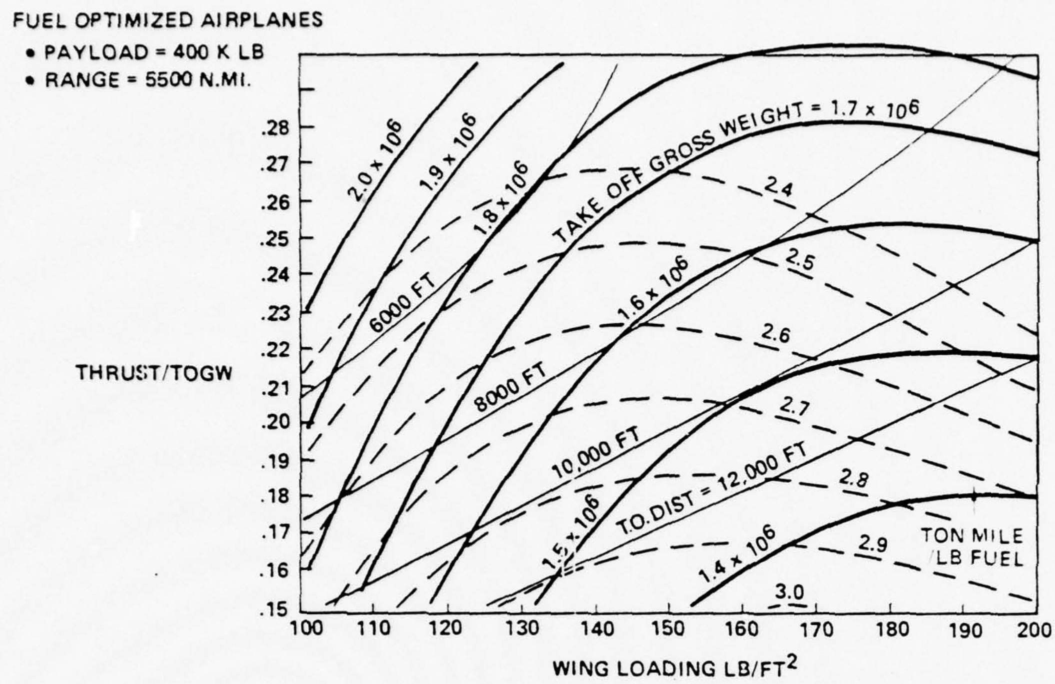


Figure 2-8 Composite Design Chart Fuel Optimized

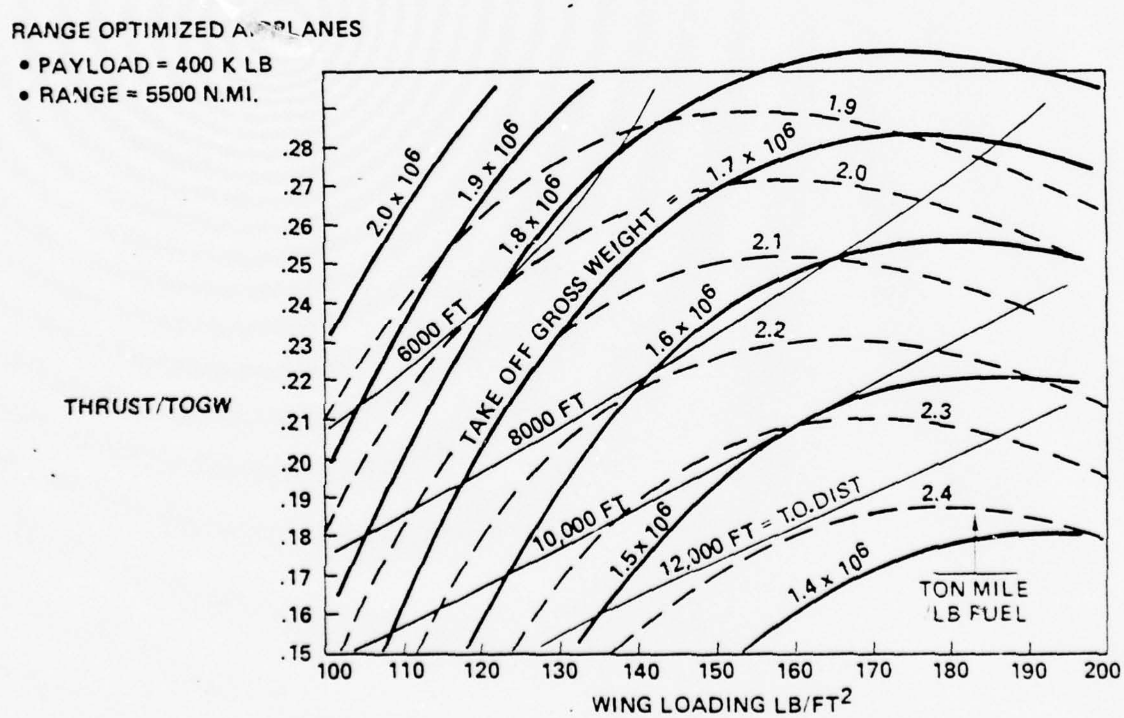
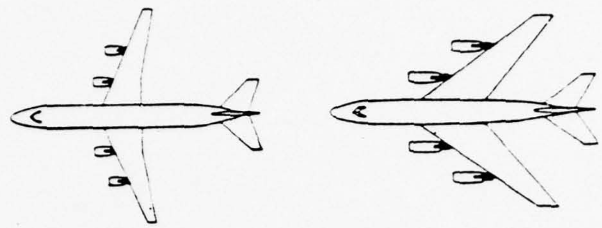


Figure 2-9. Composite Design Chart Range Optimized

The independent variables selected for analysis were the aspect ratio (AR), Wing Sweep (Λ) thickness ratio (t/c), wing loading (W/S), and thrust-to-weight ratio (T/W). The range of variables studied is shown in Figure 2-10.



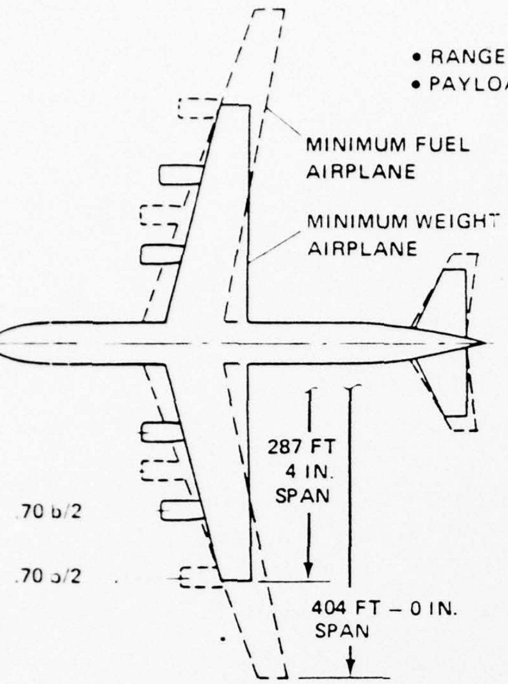
AR	8	$\longrightarrow$ 14
Λ	10	$\longrightarrow$ 30^{o}
t/c	.10	$\longrightarrow$ .15
w/s	100	$\longrightarrow$ 200
T/w	.15	$\longrightarrow$ .30

TECHNOLOGY VARIABLES

Δ OW
Δ RANGE FACTOR

Figure 2-10. Parametric Variations

An interesting comparison can be made of the effects of optimizing for fuel, or range, by comparing the planforms of the two designs as is done in Figure 2-11. Also shown are the design characteristics of the configurations. In general, the fuel efficient design emphasizes parameters which make up the Range Factor; i.e., M, L/D and SFC. Thus, the fuel efficient designs have higher sweep, higher aspect ratio and larger wings to improve the L/D. However, because of the larger wings the operating weight is substantially greater than the minimum weight designs. Consequently, although savings in the fuel costs could be antitipated, they would be achieved at the expense of increased acquisition costs.



• RANGE — 5,500 N.MI. • FIELD LENGTH – 8,000 FT
• PAYLOAD – 400,000 LB

	MINIMUM WEIGHT 1044–010	MINIMUM FUEL 1044–011
TOGW	1,600,000	1,700,000
OW	631,000	812,500
W/S	155	125
T/W	.23	.202
WING AREA	10,300	13,600
SWEEP C/4	10.9°	17.9°
AR	8.0	12.0
t/c AVE	.10	.10
HORIZ. AREA	2,260	2,880
V_H	.80	.82
VERT. AREA	1,240	1,770
V_V	.061	.047
ENG THRUST (EACH)	92,000	85,900

Figure 2-11. Comparison – Minimum Weight vs. Minimum Fuel Transports

Design points were generated for a spectrum of mission conditions encompassed by the design points specified in Figure 2-2, specifically for payloads of 200,000 lbs., 400,000 lbs., and 600,000 lbs. for fuel and range optimized designs. These results are shown on Figure 2-12. Also shown are line of constant fuel utilization showing the interesting fact that an optimum gross weight does exist which maximizes fuel efficiency at a particular design range, however, that optimum is very weak.

An interesting comparison of the parametric designs can be seen, as in Figure 2-13, where lines relating the optimum designs to lines of constant gross weight are shown. Thus in the region between the maximum range and the maximum ton mile/lb. of fuel lies a family of compromise designs which may be of interest.

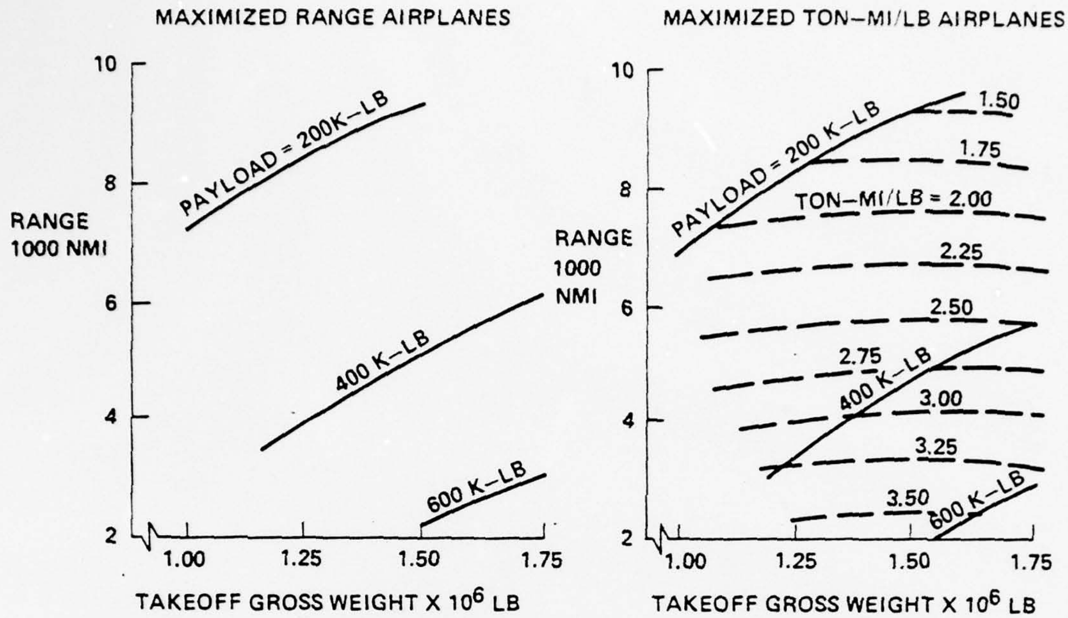


Figure 2-12. Design Sensitivities

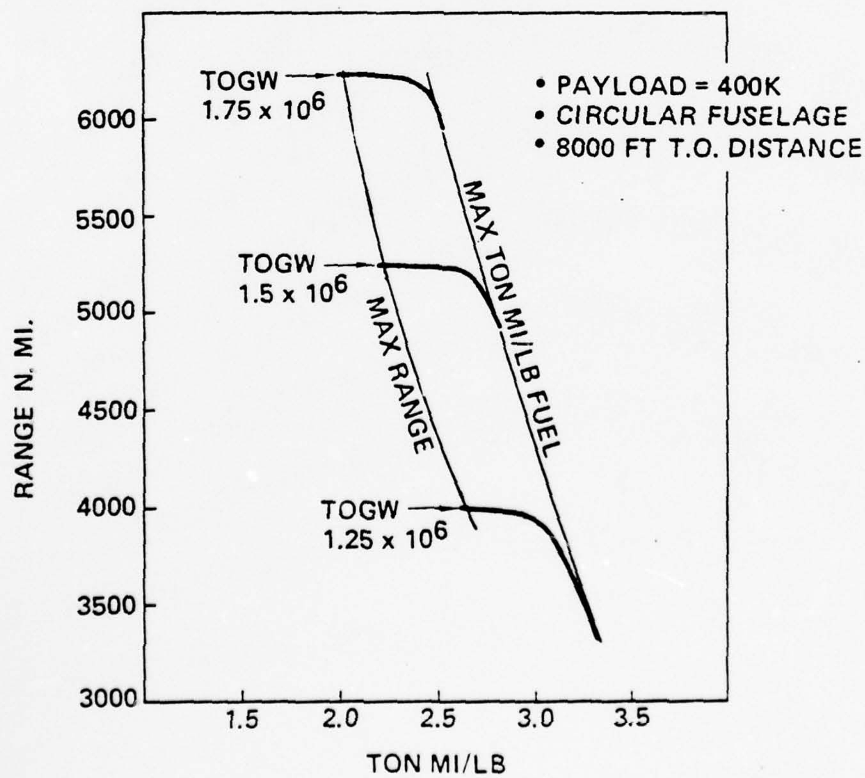


Figure 2-13. Range and Fuel Efficiency Relationships

As an example, consider the line where TOGW = 1.5 x 10^6. Additional fuel efficiency can be achieved with little cost in range until the region close to the fuel efficient design is reached. This occurs because of the limitation which was imposed on aspect ratio of AR $\leq$ 12. As shown in Figure 2-14, the increase in fuel efficiency is obtained by increasing aspect ratio until the limit of 12 is reached, after which the wing loading is decreased in order to improve the ratio of wetted area to wing area and hence, the L/D. At wing-loadings less than 120, the onset of the square cube law and the resulting wing weight make further increases in wing size counter productive, Figure 2-15.

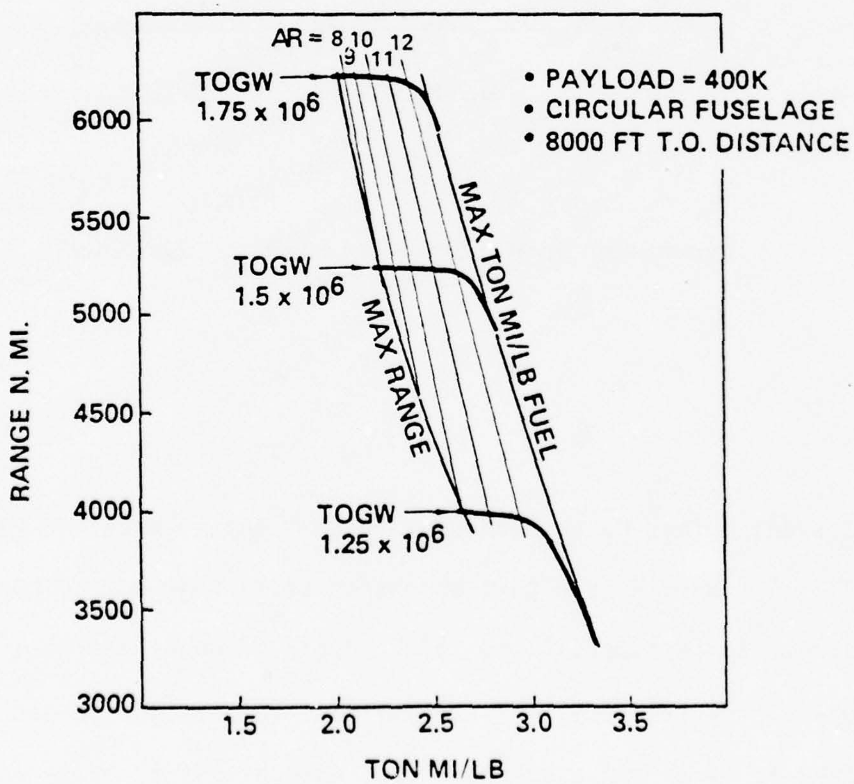


Figure 2-14. Range and Fuel Efficiency Relationships — Aspect Ratio

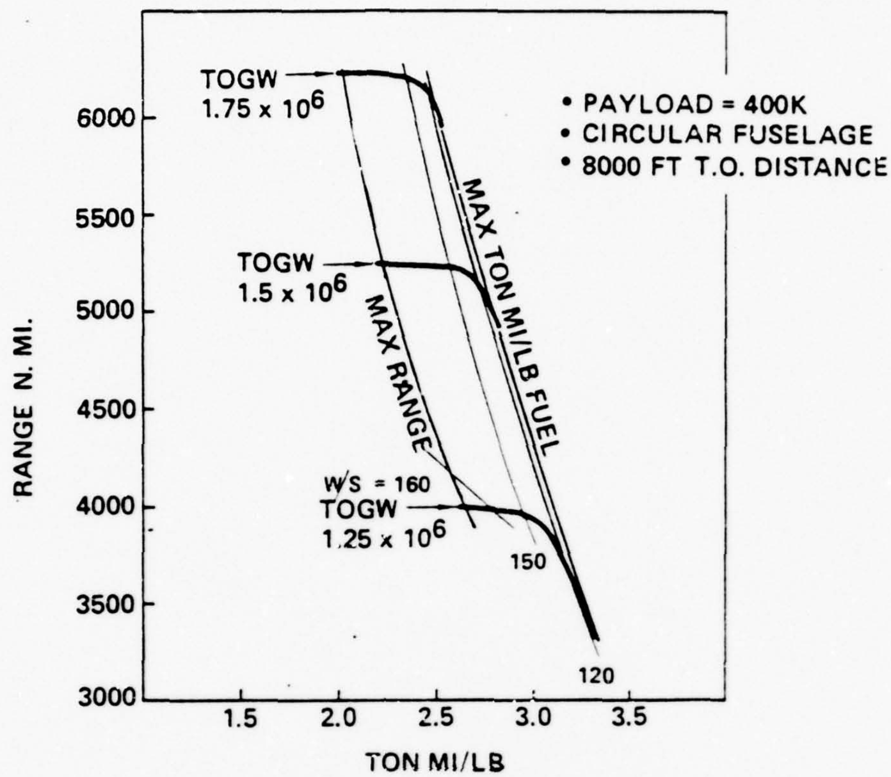


Figure 2-15. Range and Fuel Efficiency Relationships – Wing Loading

A sensitivity analysis of the impact of technology on design characteristics was made based on the parametric reference airplane, Figure 2-7. The results of this analysis are shown in Figure 2-16, and illustrate the impressive gains which can be achieved by improvements in structural weight reductions. The reference value of 1.0 in each case represents the level of technology implicit in the reference design.

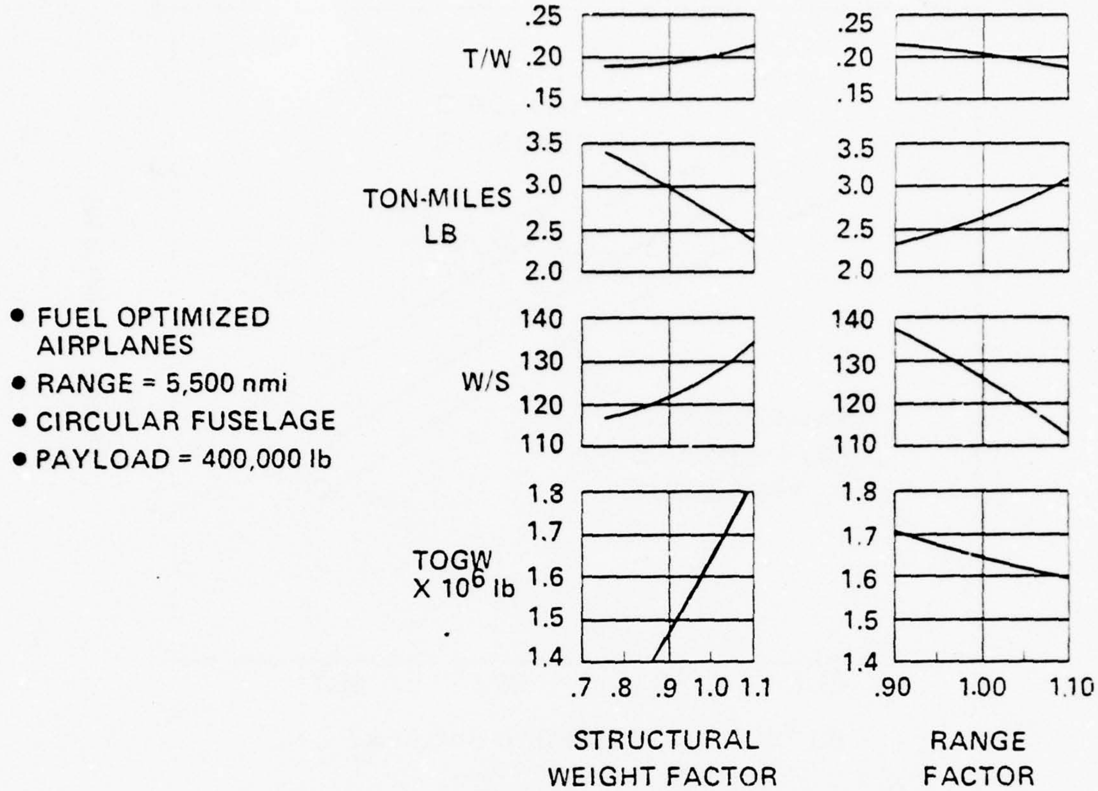


Figure 2-16. Technology Impact on Design

A parameter of great significance in the design of commercial freighters is the payload density at which airfreight moves, and which varies from 7.5 to 10 lb/ft^3. Figure 2-17 shows the impact of changing the design density for bodies of constant cross section by increasing the body length. Also shown are the effects of going to fuselage cross sections which deviate from the circular. Little difference is seen between the three and four lane configurations at this level of analysis.[1] The design cabin pressure differential was 4.5 psi which is sufficient for a cabin altitude of 18,000 feet. Decreases in cabin altitude would tend to favor the three lane configuration as will be shown in Section 3.0.0.

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[1]Analysis in greater detail in Section 3.0 revealed some differences.

Aero		
	Advanced hi speed airfoils	ΔM_{crit} + .01
	Advanced aero des methods	$\Delta(L/D)_{cr}$ + 4%
Propulsion		
	New eng	ΔSFC - 8 to 12%
	Elec fuel control	ΔSFC - 2%
	Nacelle Aero integ	ΔSFC - 1%
	Eng - Nac struc integ	ΔSFC - 1%
Structure		
	Active controls	-9% wng box wt
		-20% h-tail area
	Materials	-5% str crit wt
		-20% control surf wt
	Structural arrangement	-5% str crit empenn wt
		-12% body wt
		-5% cab wt
	Adv design meth	-2% struct wt
Mech/Elec Systems		
	ECS/avionics cooling	ΔSFC - 3% (Ref A/C)
	Carbon brakes	Δwt = 2000 lbs (Ref A/C)
	Integ actuators	Δwt = 3000 lbs (Ref A/C)
	Hi-press hydraulic	Δwt = 3000 lbs (Ref A/C)

Figure 2-18. 1985 Technology

The baseline engine, Reference 3, will provide an improvement in the SFC over the existing family of high-bypass engines and is included in the design base. Improved engine and nacelle integration will reduce installation losses by a few percent.

Significant gains appear possible in the area of improved structural design, both in the area of reduced criteria, due to the application of active controls such as a load alleviation; and also in the use of new materials such as pure alloys, and composite structure for control surfaces. Advanced design methods can also be used to reduce the impact of inefficient structural arrangement.

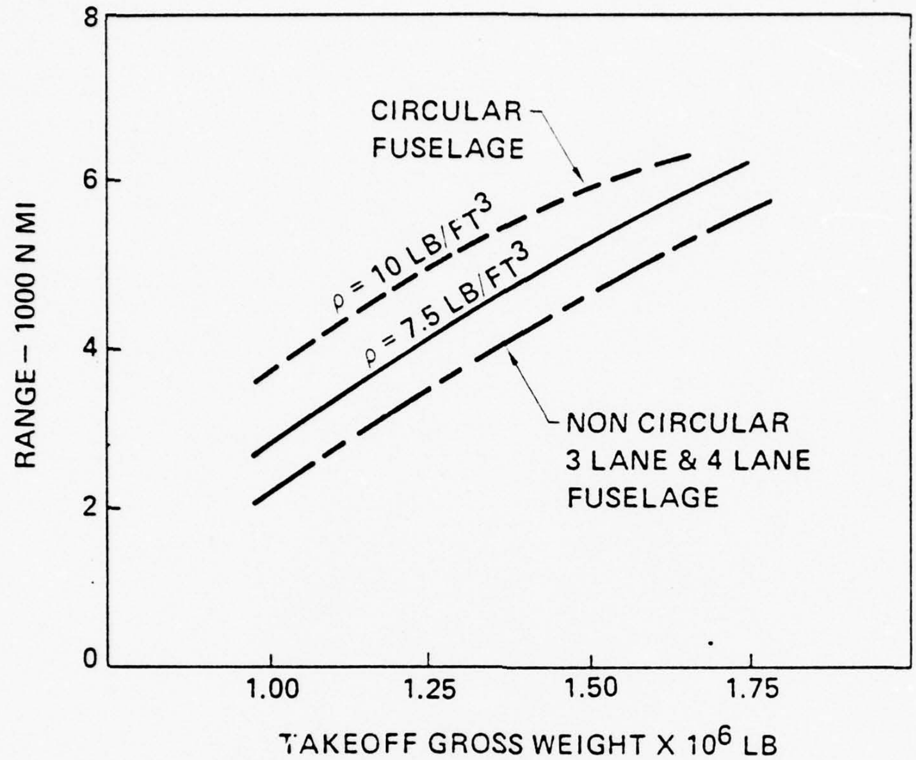

Figure 2-17. Payload Density and Body Shape Effects

Concurrent with the parametric design effort, an assessment of advanced technology was conducted to: (1) establish a 1985 technology baseline, and (2) to identify advanced technologies compatible with a 1990-2000 IOC. The technology assessment is treated in detail in Section 3.0.0, but for convenience a summary of the 1985 baseline is presented in Figure 2-18.

In the area of aerodynamics, the continuing improvement in advanced airfoil technology made possible by improved potential flow solutions, which in turn is made possible by a high speed computer, will provide an additional increment critical mach number, and will also allow for better inviscid/viscous flow solutions of wing body combinations to give an improved cruise L/D.

Advanced mechanical and electrical systems are envisioned which can provide significant savings in weight. Actuators utilizing high hydraulic pressure systems and integrated motor/actuator systems can provide weights reductions from 1 to 20 percent.

In summary, the technological base projected for 1985 provides for an estimated improvement in range factor of 17% and a reduction in operating weight of 7% over current wide body technology. Based on the 1985 technology baseline, a family of parametric designs were generated utilizing the reference design of Figure 2-7 for maximum range and minimum fuel. A representation of those designs is shown in Figure 2-19 for a payload density of 7.5 lb/ft^3.[1] For comparative purposes, the C-5 design point is also shown for a gross weight of 732,000 lbs.

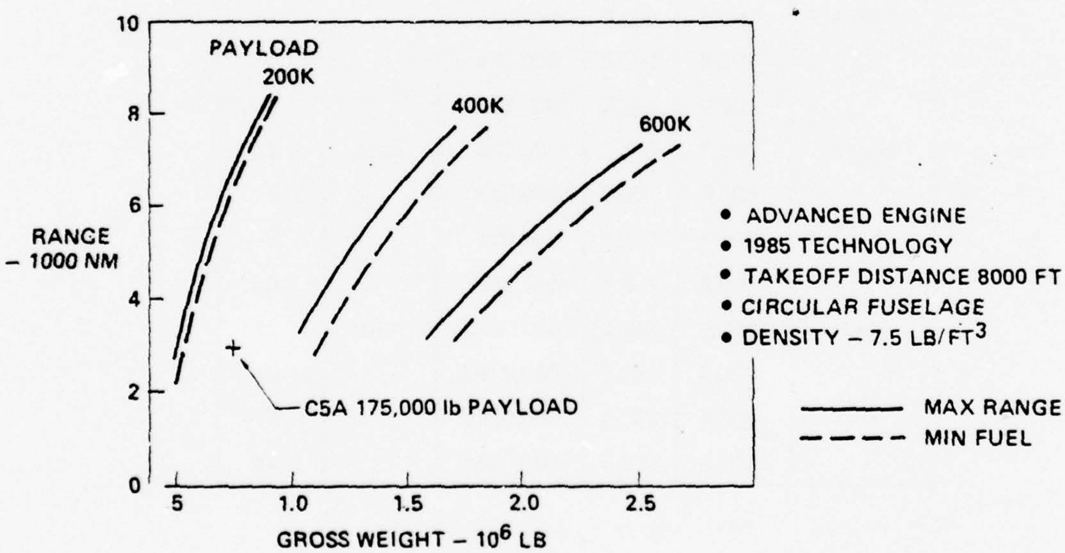

Figure 2-19. Design Solutions – Phase I Reference Configurations

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[1] Gross density, not including the tare weight and volume of the 8' x 8' x 20' containers.

$$\frac{P}{gross} = \frac{Payload}{8 \times 8 \times 20 \times \text{number of containers}}$$

2.2.4 Parametric Design Matrix Summary

Detailed design descriptions for each of the design matrix elements were generated using the CAD system. A summary of the results is shown on Table 2-1. The complete design summaries are shown in Appendix B.

Table 2-1. Parametric Design Matrix Circular Cross Section. JP Fuel. 1985 Technology

Design number	Payload	Takeoff gross weight	Design range (nmi)	Takeoff distance (ft)	Optimized for
1	400K	1,070,000	3,600	8,000	Range
2		1,310,000	5,500	8,000	
3		1,590,000	7,200	9,000	
4		1.185,000	3,600	8,000	Fuel
5		1,450,000	5,500	8,000	
6		1,680,000	7,200	9,000	
7	200K	540,000	3,600	8,000	Range
8		655,000	5,500	8,000	
9		580,000	3,600	8,000	Fuel
10		700,000	5,500	8,000	
11	600K	2,050,000	5,500	8,000	Range
12		2,475,000	7,200	9,000	
13		2,250,000	5,500	8,000	Fuel
14		2,640,000	7,200	9,000	

3.0 Preliminary Design and Advanced Technology Assessment

- Baseline Mission Definition
- Baseline Configuration Selection
  - Body Cross section
  - Wing planform
  - Landing gear
- Design Validation
- Advanced Technology Assessment
  - Aerodynamics
  - Structures
  - Propulsion
  - Systems

3.1.0 INTRODUCTION

The preliminary design and advanced technology portions of this study took place following the parametric analysis, Phase I. The preliminary design task consisted of the baseline mission definition, the baseline configuration selection and the baseline design validation.

The advanced technology assessment task was structured to identify design and technology innovations available after 1985 which would be cost effective contributions to the baseline configuration.

3.2.0 BASELINE MISSION DEFINITION

The baseline mission selected represents a challenge from the point of view of airplane design. The mission requirements are shown on Figure 3-1. The radius mission was selected to represent requirements to perform a logistics mission in an environment where fuel was not available or was of critically short supply. Some justification for this exists in the experience of the Arab-Israeli war of 1973 when the U.S. Military Airlift was required to take on as much fuel as it offloaded in payload.

The payload size is compatible with the payloads generally considered for international commercial freighter designs.

Radius/equivalent range	3600 nmi/6200 nmi
Payload	400,000 – 0/400,000
Field length	8,000 ft (MIL-5011A)

Figure 3-1 Baseline Mission Definition

3.3.0 BASELINE CONFIGURATION SELECTION

The selection of the baseline configuration involved not only the question of the definition of components, but also a definition of the criteria upon which the selection was to be made. The parametric studies had examined the question of the impact of design criteria on the configuration, specified by maximizing range and fuel efficiency.

Specifying the gross weight and the fuel utilization is a manner of relating system costs to operations cost, through the relationship of gross weight and the fuel utilization, without becoming involved in the complexities of a cost model.

In the ARES CAD, however, the capability exists to specify with great freedom the figure of merit upon which the configuration is optimized and subsequently selected. Specifically, the capability was available to couple the design process to the life cycle cost model, and the formula for calculating direct operating costs for a commercial transport. Thus, it was possible to relate the cost and design directly through the design and optimization process.

Of particular interest was the selection of a body cross section which would provide the most cost effective means of providing the floor area required for military loading, while also providing the volume necessary for low density commercial containerized cargo. The three cross sections considered are shown in Figure 3-2. It was also desirable to evaluate the effect of cabin pressure on the body weights.

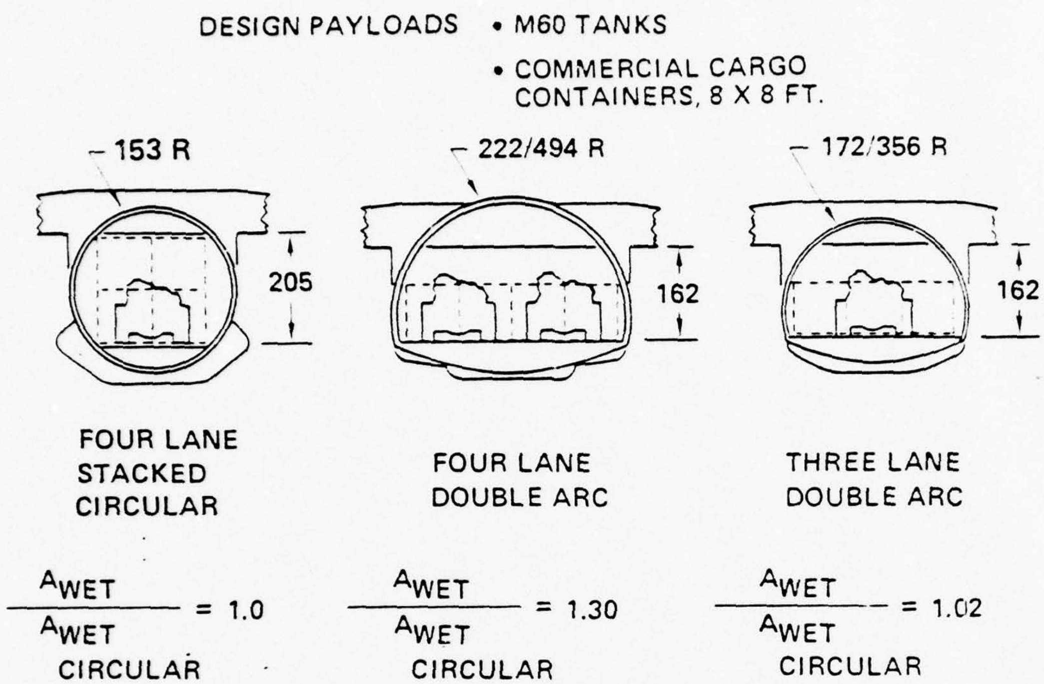

Figure 3-2 Fuselage Cross-Section

The planform characteristics were specified by t/c, Λ_E, and AR, and the sizing by identifying the W/S and T/W, which, when constrained by 8,000' field length, provided a minimum cost design.

The parametric analysis had indicated an airplane on the order of 1.5 million lbs gross weight would result from the mission criteria specified for the Baseline Mission. For this reason, there was concern that a landing gear could not be designed with sufficient flotation to properly distribute the loads to the existing runway pavements and provide for sufficient durability of runway pavements. For a criteria, the LCN and pavement stress levels of the 747 were selected.

3.3.1 Body Cross-Section

The body cross section selection is of critical importance in selecting a military transport which can operate at a minimum cost in performing its mission of transporting military equipment. Several ground rules were specified prior to the definition of the cross section. First, the configuration must permit drive-through capability, with fore and aft ramps; second, it must be designed to a cabin pressurization level of 4.5 psi, corresponding to a cabin altitude of 18,000 feet while at cruise altitude; third, it must have a floor strength equivalent to that of a C-5A and be capable of carrying a M-60 main battle tank at 2.5g.

The U.S. Army Mechanized Division was selected as being typical of the average military payloads which would be required.

The characteristics of the cargo compartment determine the ability to transport efficiently the design payload as well as influence other design considerations such as floor height, tail arm, and landing gear arrangement. A parametric study was undertaken for the JP configurations to establish the relationships between these design considerations and cargo carrying efficiency. The following questions were addressed:

1. Do particular box geometries lend themselves to greater packaging efficiency?

2. Are loading efficiency results sensitive to type of payload? For example, would conclusions be different if an Infantry Division were assumed in place of an Armored Division?

3. Do military packaging charcteristics have to be compromised to accommodate good civil cargo loading characteristics?

4. Are desirable cargo box geometries compatible with other design requirements--such as floor height and landing gear arrangement?

A computerized loading model was used to provide a parametric description of loading characteristics of candidate configurations to provide information to address these questions, Reference 4. The loading model results were combined with design results of the ARES model to relate loadability to configuration changes. These results were in turn used to define fleet Life Cycle Cost (LCC) for each configuration. The LCC results were then used to compare the relative merits of the various cargo box concepts.

3.3.1.1 <u>Military Requirements</u>

Figure 3-3 summarizes vehicle dimensions for loading of armored, mechanized, infantry airmobile and airborne divisions. Vehicles peculiar to the critical support increment and the CH47 helicopter from the Airmobile division are not included.

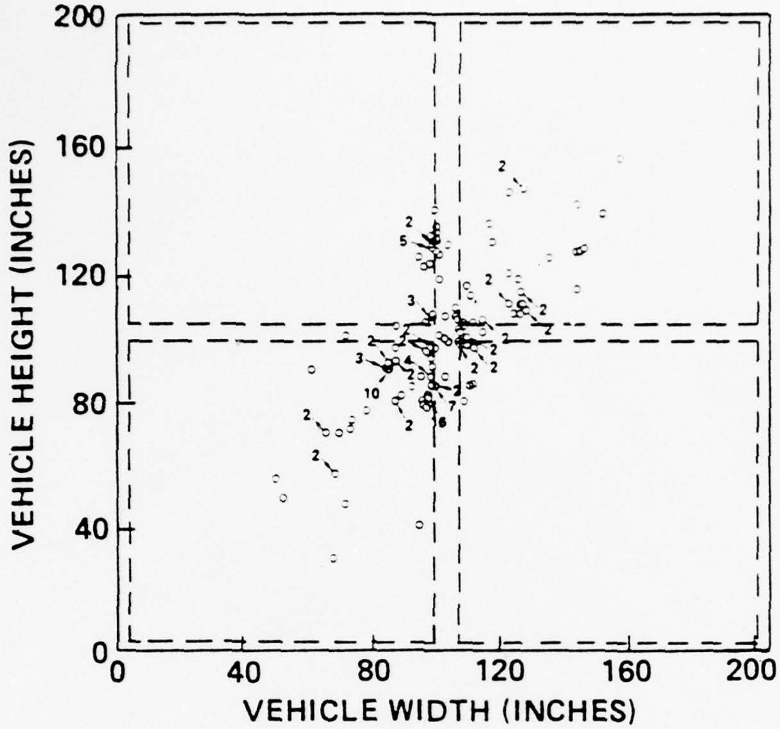

Figure 3-3 Army Vehicle Cross Section Distribution

The majority of vehicle types are concentrated around a width of 100" and height of 100". The vehicle with greatest cross section dimensions - an Armored Vehicle Bridge Launcher (AVBL) - has height/width dimensions of 159 inches and 158 inches respectively. The 159 inch height along with loading clearances is used to define ceiling height for candidate cargo compartments. Since the narrowest floor considered in this study is that associated with two 8 feet cargo lanes (approximately 200" overall floor width) the AVBL width does not directly influence cargo floor width selection.

Overall vehicle width distribution for a division, however, does influence the floor width selection. If, for example, a large percent of the vehicles in a division of interest were AVBL's, a floor with some multiple of the 158" AVBL width would provide the best floor loading efficiency.

Figure 3-4 and 3-5 summarize the width distribution of Mechanized and Infantry Divisions when the number of each type of vehicle is considered. The Mechanized Infantry Division is seen to have a wider average width than the Infantry Division and should therefore present greater difficulty in achieving efficient floor loading.

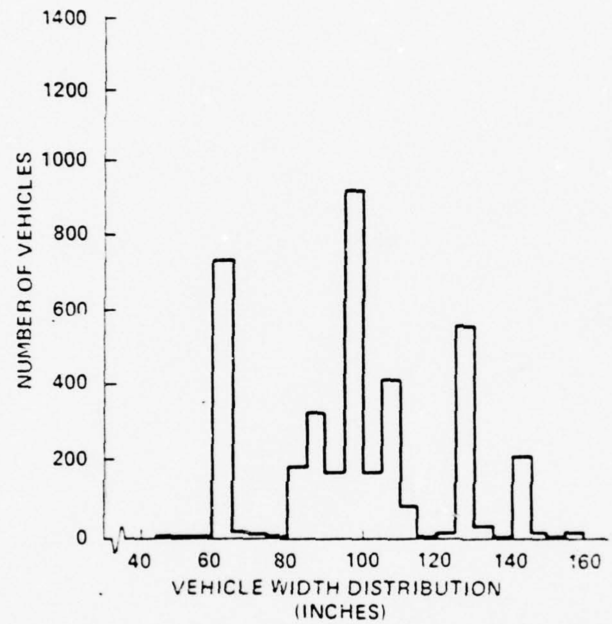

Figure 3-4 Vehicle Width Distribution Mechanized Infantry Division

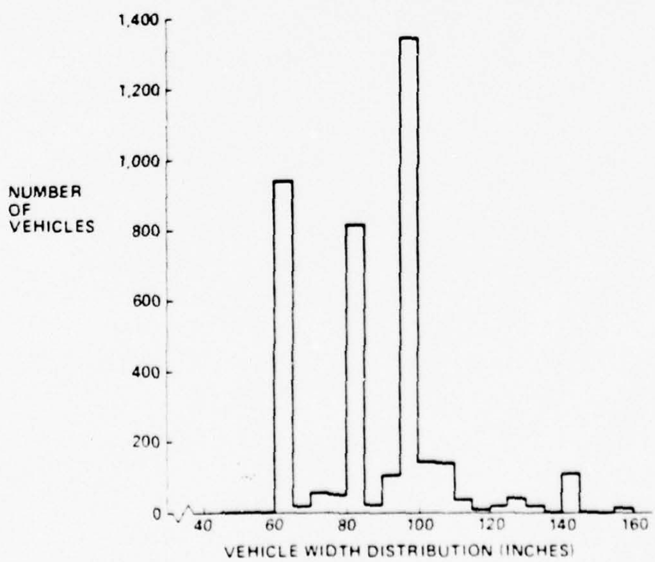

Figure 3-5 Vehicle Width Distribution Infantry Division

3.3.1.2 Civil Cargo Requirements

Desirable cargo box characteristics for a civil transport of 400,000 lbs. payload do not match those for a 400,000 lb. military payload. Floor loading requirements for 8 x 8 air/surface intermodal containers would be 60 to 80 lb/ft^2 for cargo densities ranging from 7.5 to 10 lb/ft^3 (including container tare weight). If the 8 x 8 containers are stacked, as in the case of the circular fuselage configurations, the characteristic civil floor loadings range from 120 to 160 lb/ft^2 for the 7.5 to 10 lb/ft^3 overall container densities, respectively. The military floor loadings, on the other hand, are typically 85 lb/ft^2 for an Infantry Division and 120 lb/ft^2 for a Mechanized Infantry Division. Since a density of 10 lb/ft^3 would be the more typical civil density, and the Mechanized Infantry Division the best estimate for a military payload for a NATO surge mission, a floor loading incompatibility exists between civil and military payloads for either circular or double arc configurations. The penalty associated with this incompatibility will be explored in the following analysis.

Figure 3-6 describes the floor area versus floor length for the four JP body cross section configurations. The floor lengths associated with civil cargo densities, between 7.5 and 10 lb/ft^3 are seen to range from 148' to 197' for the four lane configuration and 200' to 265' for the 3 lane configuration. Although a civil cargo density of 10 lb/ft^3 is judged to be the most appropriate for the four lane baseline, a design point was selected at 9 lb/ft^3 to satisfy the additional requirement of a floor length capable of accommodating 8 x 8 containers of 20-foot lengths (i.e., 160' floor length). Because of the container stacking used for the circular fuselage configurations, floor length requirements for civil requirements are identical to the 4 lane configuration.

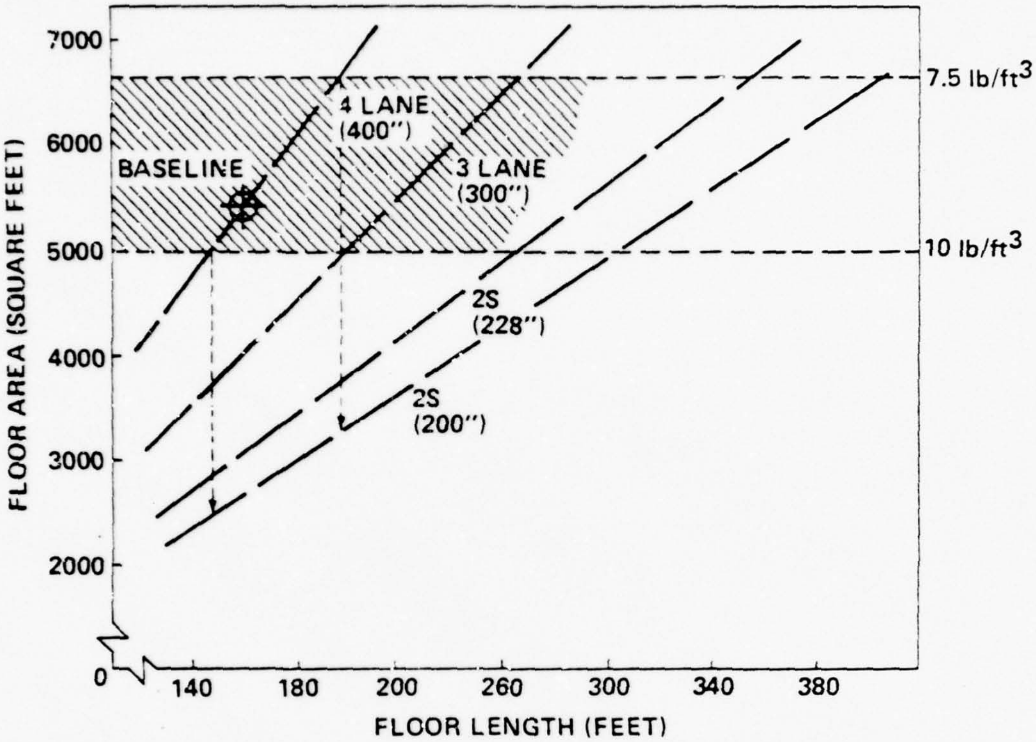

Figure 3-6 Available Floor Area versus Floor Length

3.3.1.3 Loading Analysis

The parametric loading analysis used to establish relative loading efficiency of various cargo box concepts employed the Airlift Loading Model, referred to earlier. To provide maximum coverage of geometry variables, the analysis was performed, using the Mechanized Infantry Division as a typical NATO mission payload. Vehicles were loaded without troops and to cross country weight.

Sorties required to transport the Division to Europe were used as the figure of merit. These sortie levels required to carry the Initial Support Increment (ISI) of the division are not included.

The sortie requirements for a 5,400 square foot floor area, the floor area of the baseline configuration, are summarized for various cargo floor widths in Figure 3-7. The comparison shows that the sorties required to carry one Mechanized Infantry Division is relatively insensitive to floor widths at the wide body floor widths associated with a 400,000 lb. payload. The same trend is seen for an Infantry Division.

This sortie trend changes somewhat as floor area is reduced. Figure 3-8 describes the Mechanized Infantry Division sortie trend versus floor width for various floor areas. The comparison shows a definite sensitivity to floor widths on the order of 200. The drop in required sorties at the 228" floor width is significant for the two-lane circular fuselage. As will be seen in the following discussion, this trend can help offset the floor area restrictions of circular fuselage for the military missions.

The above loading results have assumed that vehicle loading order has no priority. That is, vehicles are loaded without regard for keeping fighting units together on same sortie or consecutive sorties. Previous loading results indicate that if the unit integrity rule were applied to the loading analysis, no change would occur in study conclusions. The major effect of unit integrity loading would be to increase sortie requirement for a given case by approximately ten percent.

The overall conclusion that can be drawn from this analysis is that discrete floor widths that satisfy civil 8 x 8 container requirements can be selected for floor widths ranging from 250" to 400" without adversely affecting

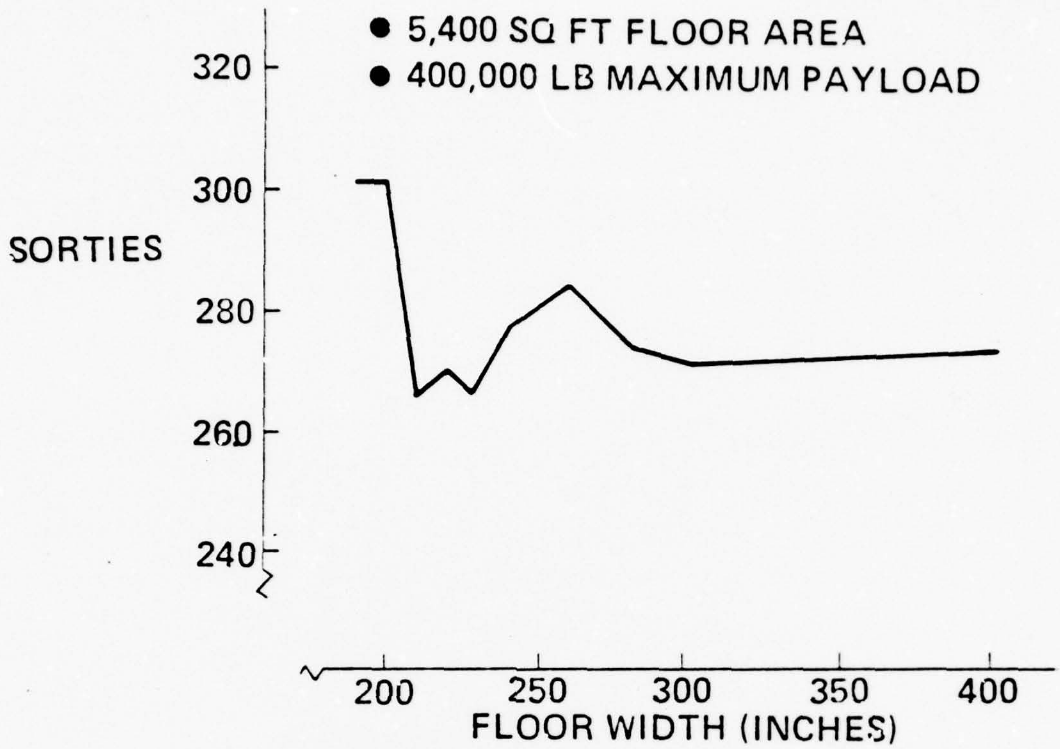

Figure 3-7 ALM Loading Results

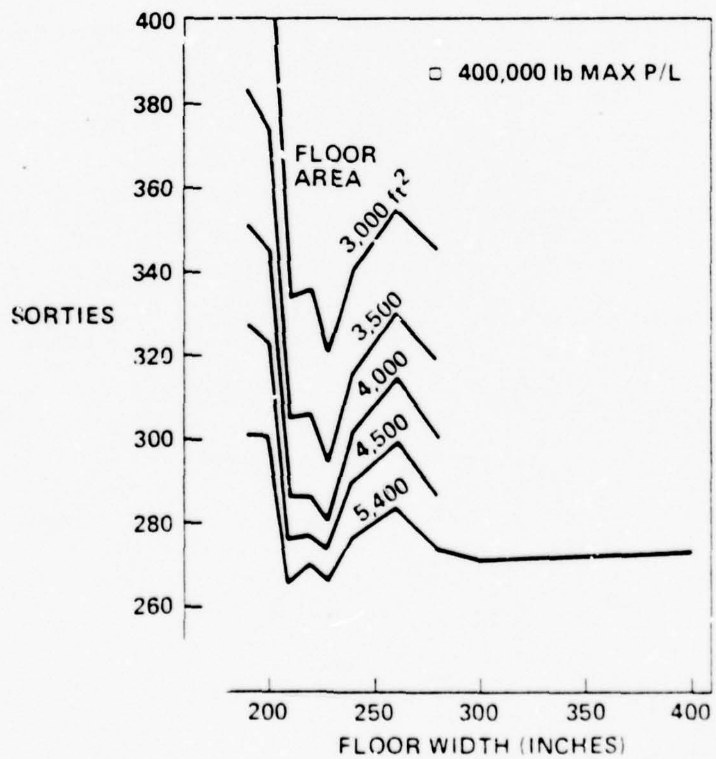

Figure 3-8 ALM Loading Model Results Mechanized Infantry Division

military loading efficiency. The 250" - 400" range corresponds to the three lane and four lane configurations. However, when selecting floor width for the two lane configurations, care must be taken in selecting a floor width greater than the 200" needed for two lanes of cargo containers. A floor width between 210" and 230" would serve as an efficient military transport and also satisfy requirement for two lanes of 8 x 8 containers.

3.3.1.4 Life Cycle Cost Implications

The loading results of the ALM analysis have been used along with the results of the parametric configuration study to evaluate cargo box concepts in terms of LCC. Figure 3-9 summarizes results of the parametric configuration study in terms of AMPR weight. AMPR weight is used to correlate configuration changes with LCC changes. The ALM results were used to adjust fleet size to account for LCC changes due to changing sortie requirements resulting from restricting or upgrading available floor area. Fleet productivity was fixed at one Mechanized Inventory Division to NATO in a given time interval. The net effect on LCC of configuration changes and fleet size changes is described in Figure 3-10. The overall result of this approach is to define the best possible floor length, in terms of LCC, for the floor width of interest.

The results of applying this LCC evaluation technique to the two lane, three lane, and four lane JP configuration are summarized in Figure 3-11. These results show similar LCC levels for the four lane, three lane and 228" two lane configurations. The 200", two lane, configuration has a substantially higher LCC level than the other three configurations. The design points that correspond to a 10 lb/ft^3 civil cargo density are seen

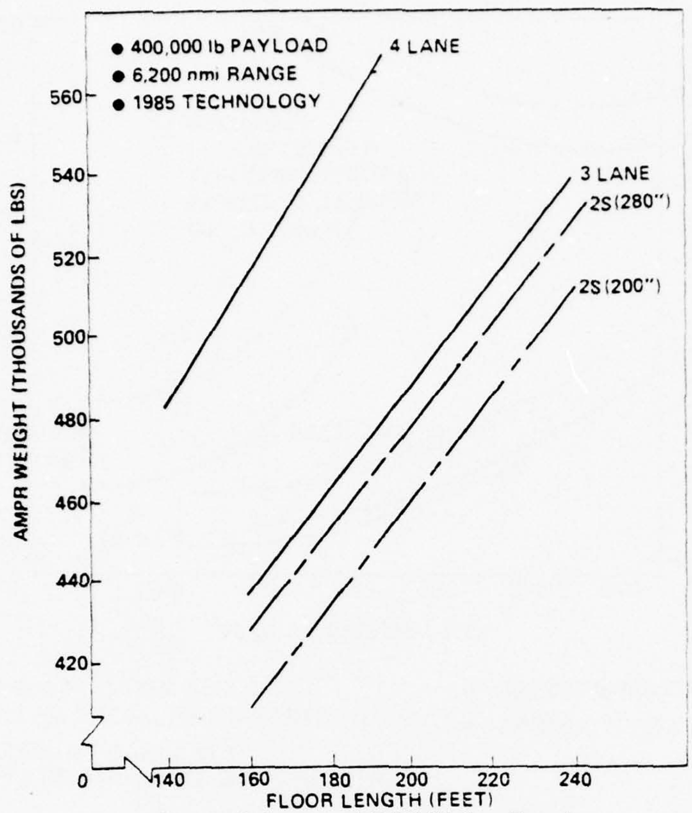

Figure 3-9 ARES Sizing Results

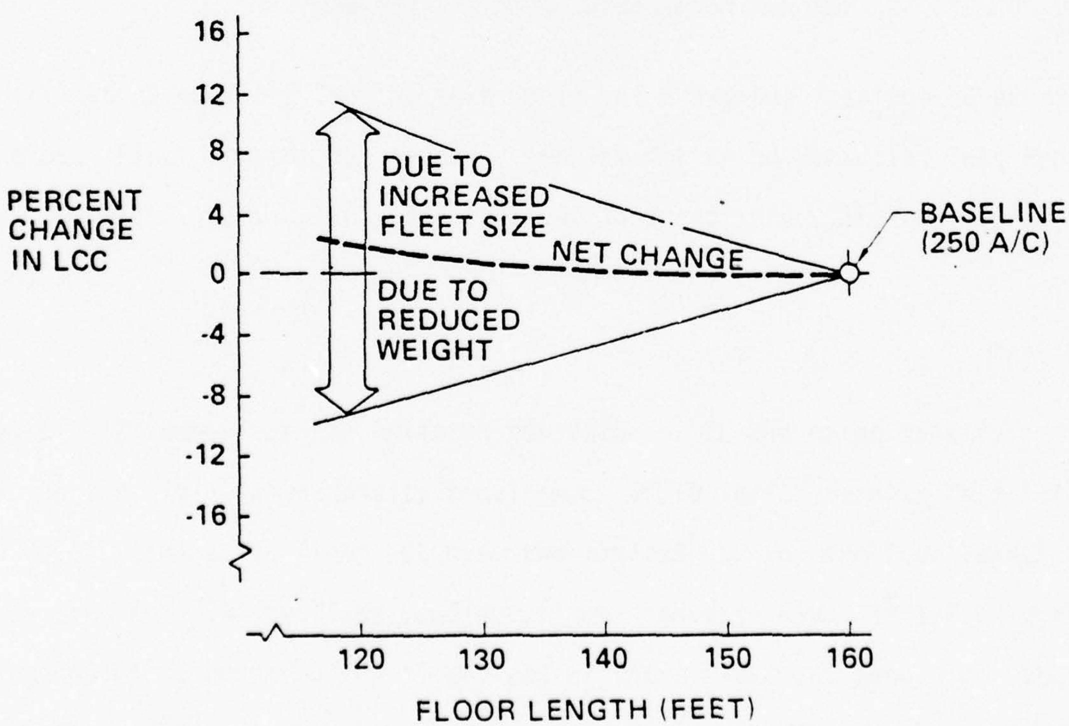

Figure 3-10 Changes in Fleet LCC with Change in Floor Length

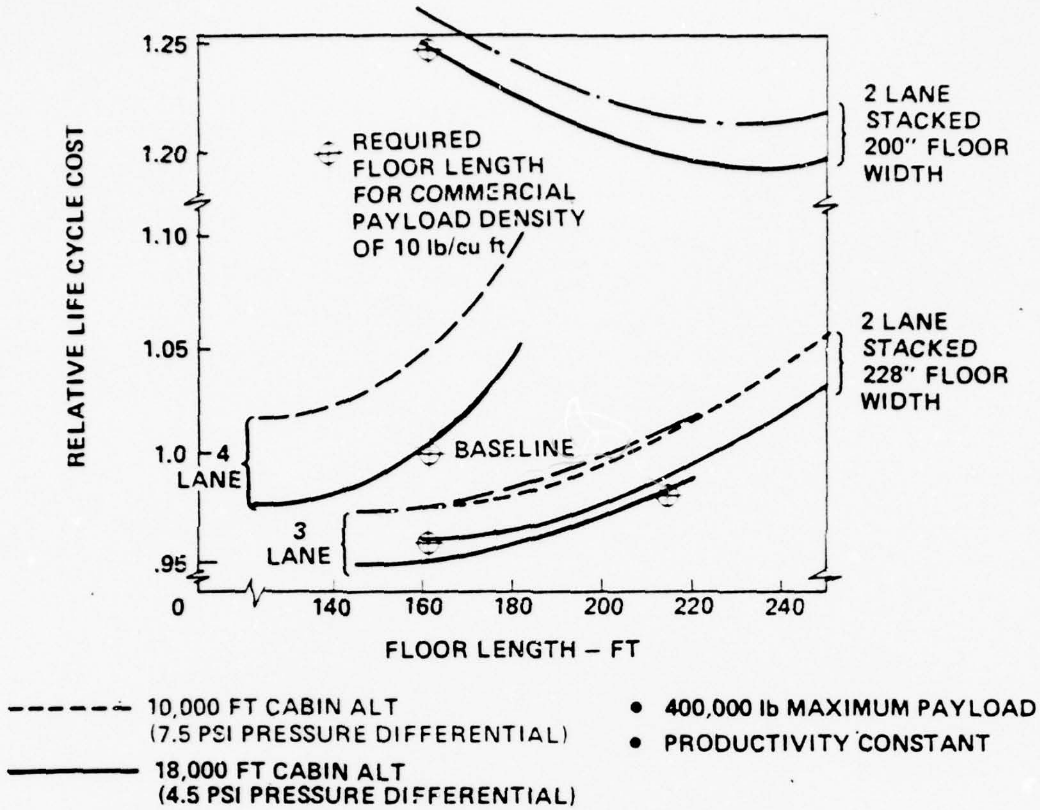

Figure 3-11 Fuselage Cross-Section Selection

to fall at cargo floor lengths that would not drive the fuselage to excessive lengths. Floor lengths may become a problem for cargo densities less dense than 10 lb/ft^3, particularly for the four lane and three lane configuration.

Figure 3-12 summarizes the relationship between LCC and cargo density. Meeting the civil cargo density requirement of 10 lb/ft^3 results in a 1.1, 2.0, 0.6% increase in fleet LCC over the minimum LCC for the four lane, three lane and 228" two lane configurations, respectively. If a 7.5 lb/ft^3 design point is required, civil payload of these fleet LCC penalties become 10.5, 11.0, .5% respectively.

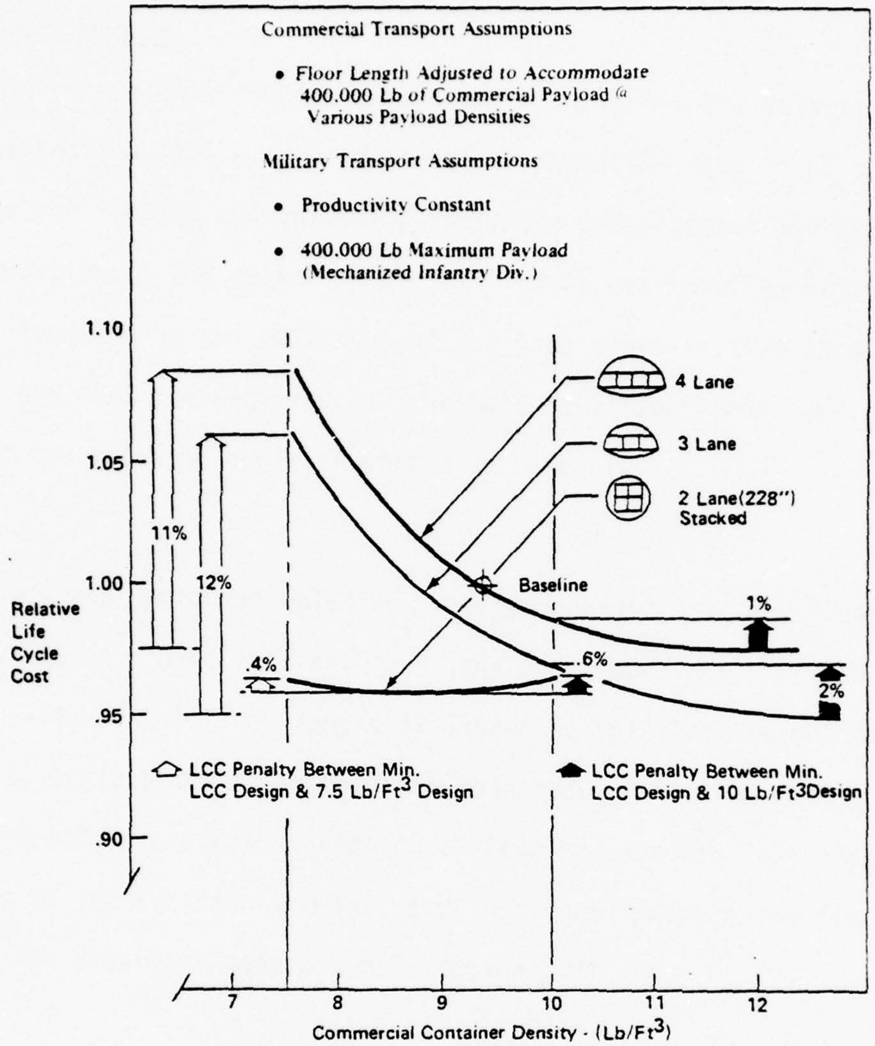

Figure 3-12 Payload Density

Based on these IADS study results, it can be concluded that no significant fleet LCC penalty is incurred when the military transport design is compromised to accommodate civil cargo at 10 lb/ft^3. Of the three configurations, the 228" two lane design appears the least sensitive to design penalties when civil cargo densities are reduced below the 10 lb/ft^3 value. Neither do the civil cargo box requirements drive the fuselage design to levels which are incompatible with floor height and landing gear requirements.

3.3.2 Floor Heights

In addition to the cost considerations which influence the selection of the body cross-section, the height of the floor above the ramp is of critical importance for loading and unloading equipment under combat and quasi-combat conditions. The selection of the landing gear configuration and length is impacted by the selection of the body cross-section through the length of the fuselage required for the necessary floor area; and consequently to the rotation angle required for take-off.

A trade study was performed relating the floor height to cargo floor length for the two lane,[1] three lane and four lane cross-sections. As can be seen on Figure 2-14, as the floor length is increased the floor height must be increased to achieve the rotation angle required. After 160 feet the four lane configuration requires an additional landing gear post per side as a consequence of the additional weight and therefore, reduces the floor height by placing the aftermost gear assembly further aft.

The significant point to be made is illustrated by the two design points which show the floor heights for the four and three lane configurations for equal floor area. The four lane configuration has a deck height which is about 3 1/2 feet lower than the three lane and substantially lower than the circular configuration.

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[1] A lane is defined as the width and height of an 8 x 8 container.

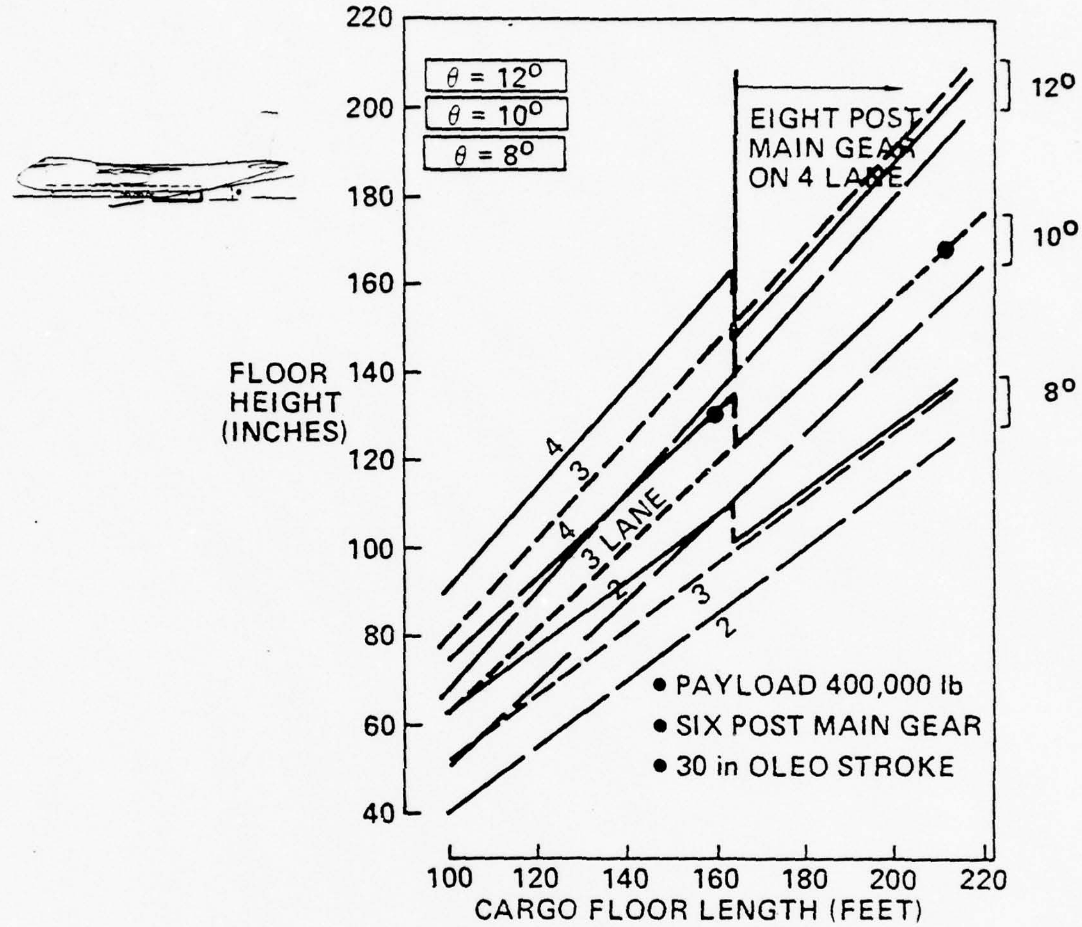


Figure 3-13 *Cargo Floor Height versus Length*

3.3.3 Wing Planform and Sizing

A comparison was made to define the implications of selecting wing size and characteristics on the basis of life cycle costs. Life cycle costs are essentially an index of peace time costs with the Operations and Support component usually based on the assumption of low utilization rates to be compatible with current MAC practice. This assumption, when integrated into the optimization process which provides the basis for planform selection, emphasizes the significance of acquisition cost rather than operating cost. This results in planform characteristics which minimize cost rather than maximizing performance.

On the other hand, direct operating costs, when used as a figure of merit, provide sufficient emphasis on block speed so as to provide for a configuration which has better performance.

3.3.3.1 Composite Design Charts

Composite design charts were constructed for the Baseline Mission showing the variation of W/S and T/W with airplane gross weight, subject to the field length constraint. Figures 3-14, 15 and 16 show those relationships for a design optimized for gross weight, life cycle cost and D.O.C.

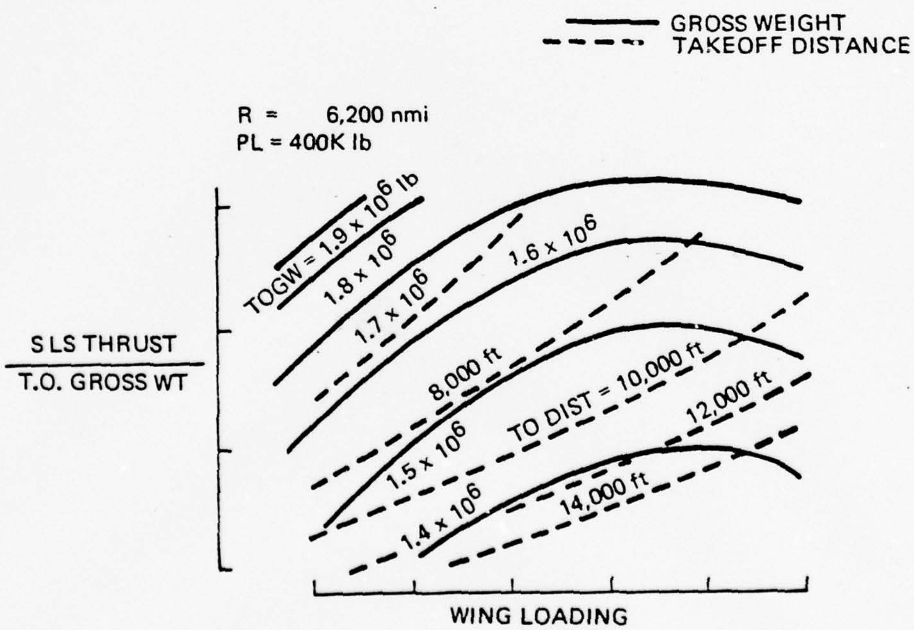


Figure 3-14. Thrust Wing Loading Design Surface Minimum LCC Designs

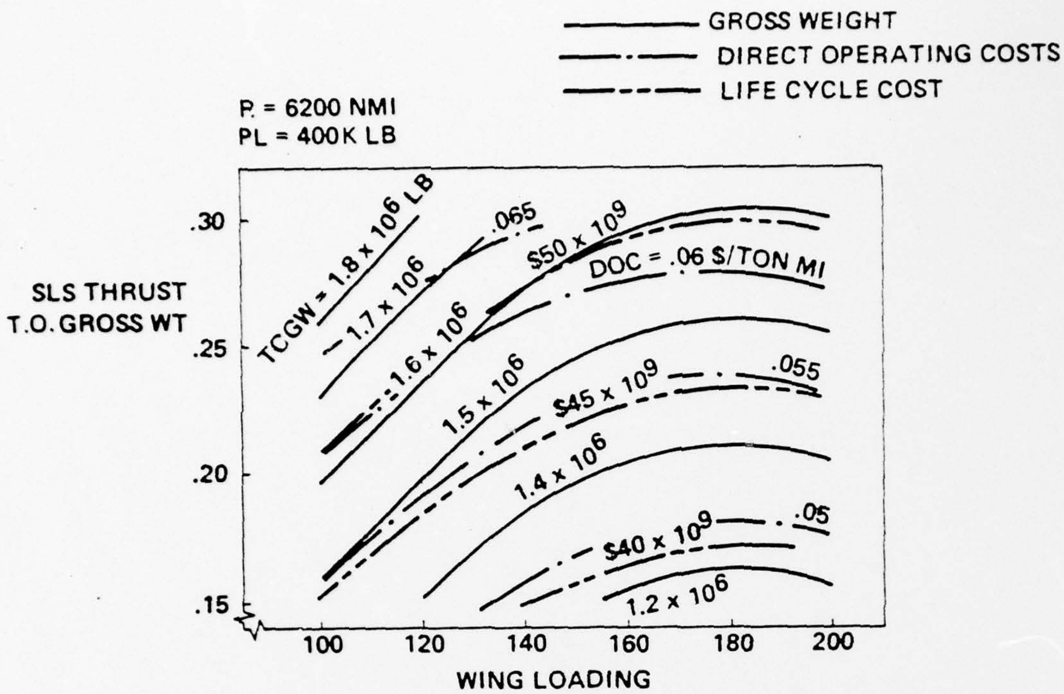


Figure 3-15 *Thrust Wing Loading Design Surface Minimum DOC Designs*

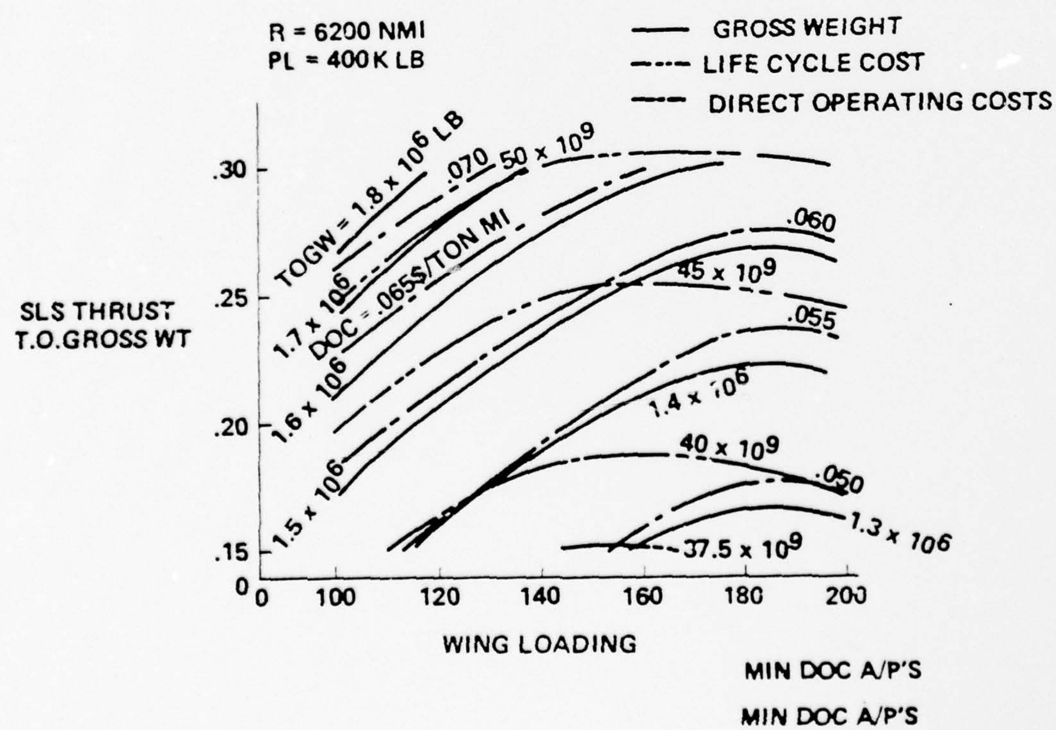


Figure 3-16 *Thrust Wing Loading Design Surface Minimum Gross Weight Designs*

Generally, as might be expected, the imposition of increasingly stringent field length requirements result in increasingly larger wing areas, higher thrust levels and higher gross weights.

A comparison between the three design criteria; minimum gross weight, minimum LCC and minimum DOC, shows the design point for minimum gross weight criteria occurring at the highest wing loading of the three criteria and producing a gross weight slightly lower than the minimum DOC designs. The minimum LCC and DOC design points occurred at wing loadings which were quite close to one another with the minimum life cycle cost design point being slightly lower. Thrust levels responded to decreases in wing loading by decreases in T/W.

During the optimization process a number of variables were left unconstrained so that the optimum combination of independent variables produced the desired maxima. In order to maintain control over the optimization process, and to better portray the strength of the trends, a trace of the design variables was constructed as a function of design range, Figure 3-17. Also shown are the variable limits which were imposed to maintain the accuracy of the design synthesis.

Of particular interest is the fact that the LCC design is as driven to the most conservative plan form characteristics: $t/C = .15$, $AR = 9.0$ $\Lambda_{LE} = 10$. This, of course, resulted in a decreased level of performance as will be shown on Figure 3-18.

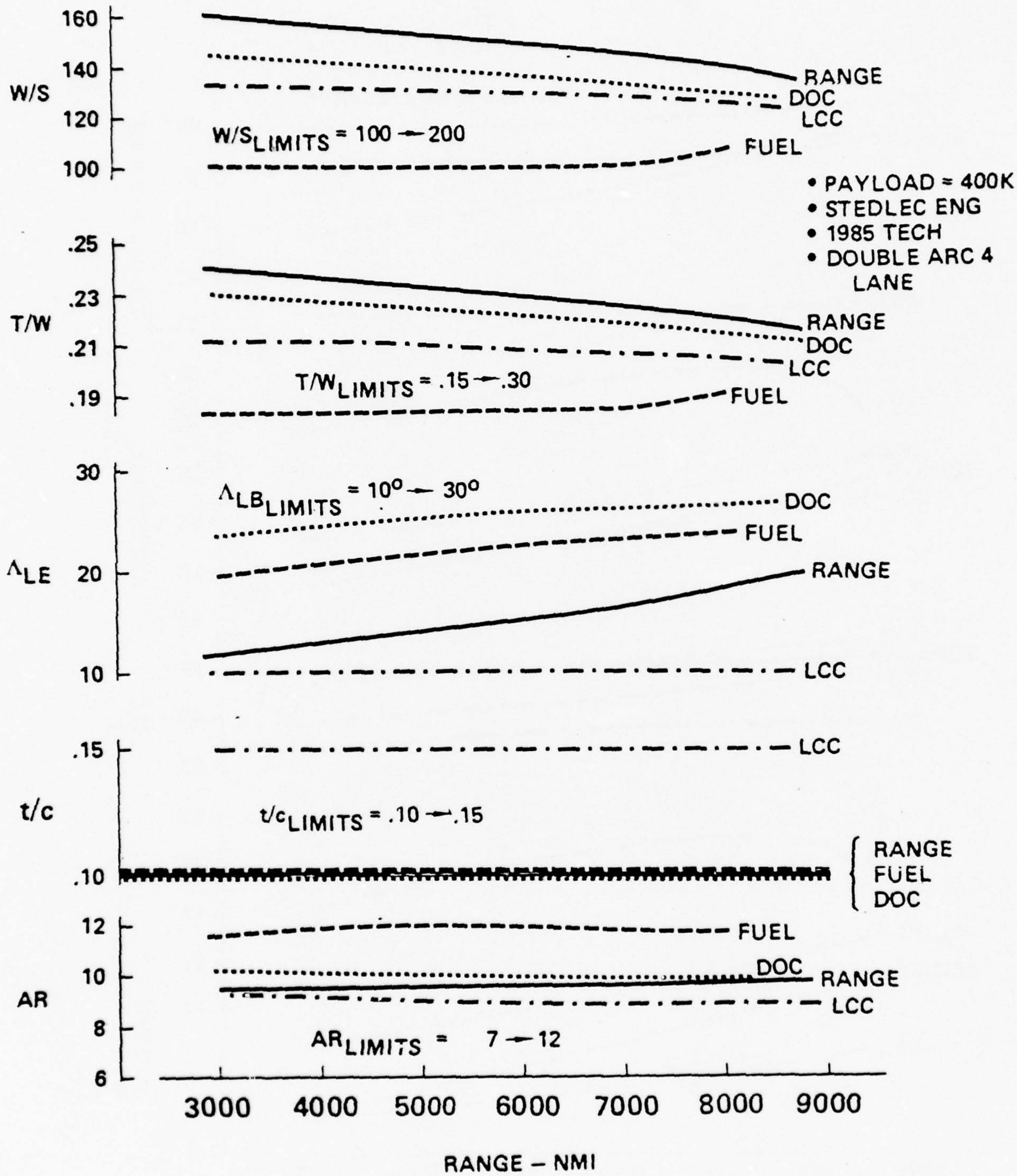


Figure 3-17 Optimum Airplane Configuration Description

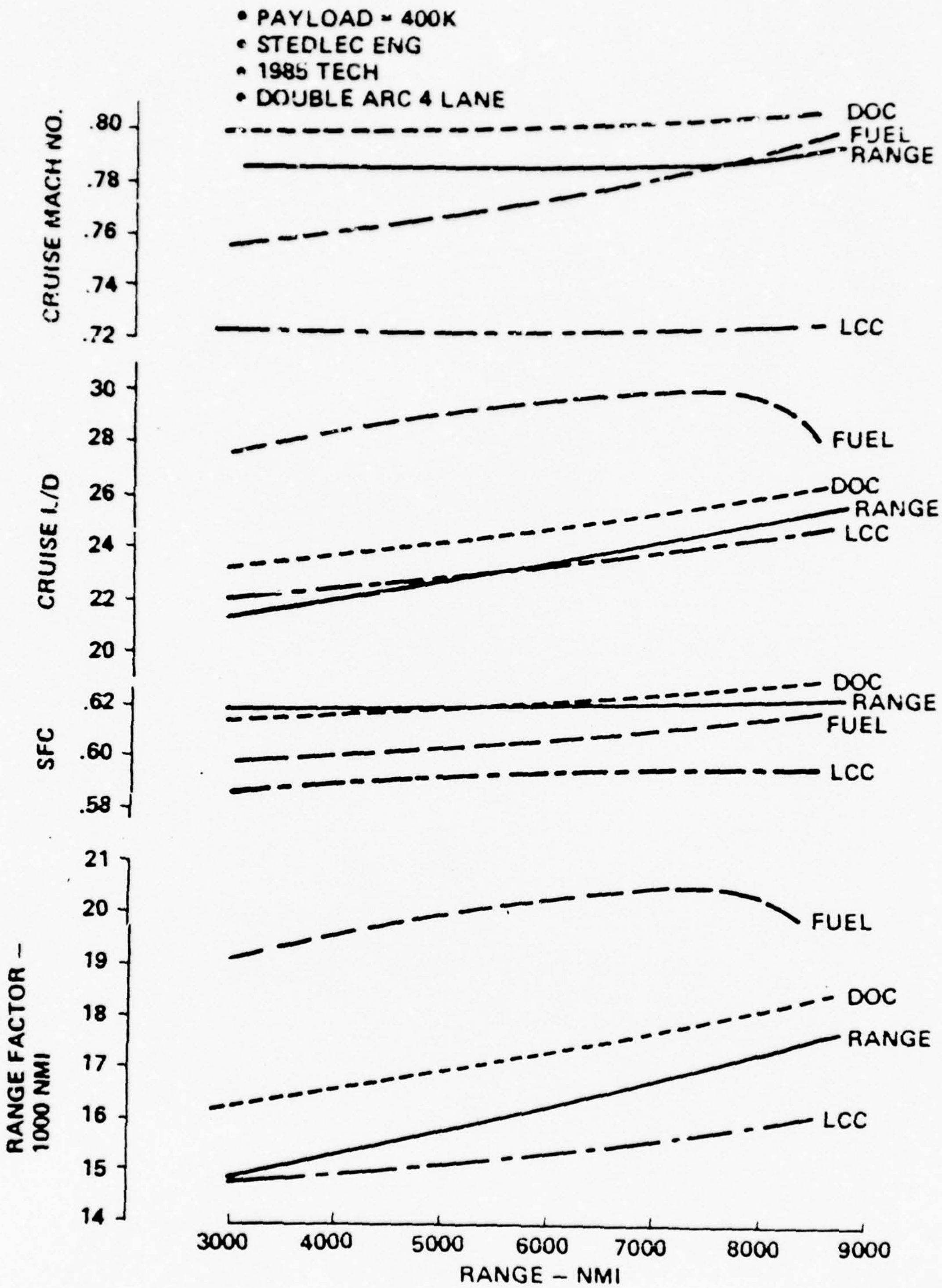


Figure 3-18 Optimum Airplane Cruise Performance

The emphasis which DOC places on cruise mach number is shown by the fact that the designs optimized for DOC have the highest sweep, lowest thickness ratio and results in the highest cruise mach number.

The wing loading trace at the top of Figure 3-17 are the fundamental data which drive the change in design characteristics as range is increased. It is significant to note the convergence of designs as range increases. At some range, between 9000 and 10,000 nmi, the maximum range design reaches the limit of its capability and increases in gross weight no longer result in increases but decreases in range. This phenomenon occurs as a result of the logarithmic nature of the range equation and the implications of the square cube law. However all designs will converge to the maximum range design at the maximum range design point for the technology base in question. The beginning of that convergence is shown in Figure 3-17 on the trace of wing loading versus range. The principle of design convergence is further illustrated in Figure 3-18 where performance characteristics traces are shown.

The cruise Mach number differences of the designs are shown to be significant in the region of interest, particularly with regard to the LCC design. The life cycle cost design, because of its lack of emphasis on the importance of operating costs results in a design which, although an inexpensive airplane to procure, is a poor performer.

The cruise L/D trace shown on Figure 3-18 is worth some examination as it is a major contributor to the range factor. As the wing loading is decreased in response to the demands of greater range, as was shown on Figure 3-17, the ratio of wetted area to wing area is also decreased thereby reducing the zero lift drag coefficient and improving L/D at cruise. The trace of Awet/S is shown on Figure 3-19 and also shows the similarity between the weight ratios of the minimum LCC and maximum range designs at lower ranges. However as the range requirement is increased, the range design, which best illustrates a compromise between the influence of range factor and operating weight, requires a better range factor than does the LCC design. This is achieved primarily by moving to increasingly higher Mach numbers at the expense of increasing the weight ratio.

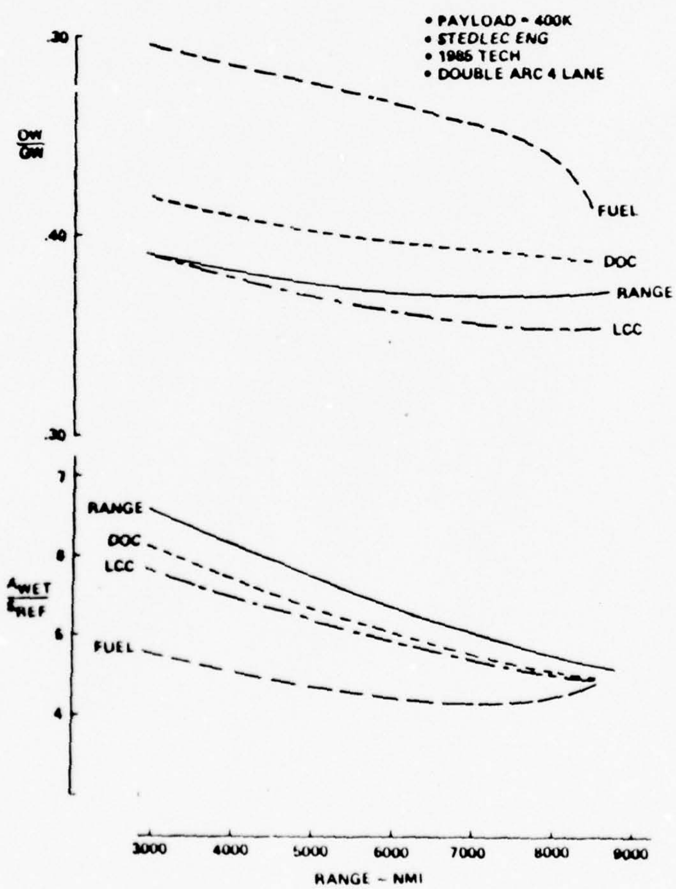


Figure 3-19 Optimum Airplane Design Parameters

The impact of design criteria was examined as it impacted gross weight, a conventional representation of design effectiveness as shown in Figure 3-20. Clearly the impact of requiring maximum fuel efficiency is costly in weight and as a result, in cost.

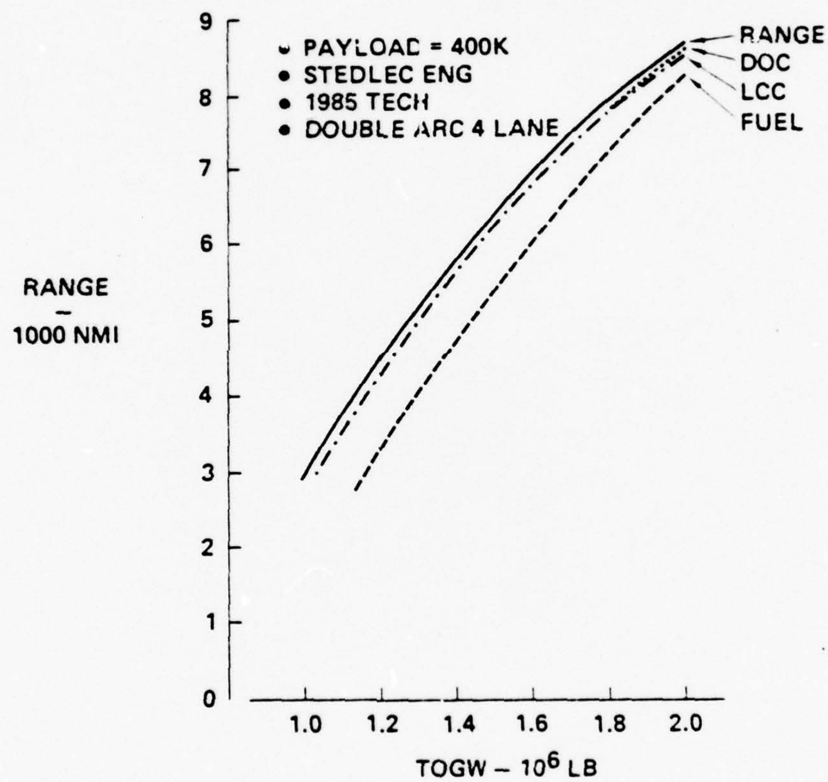


Figure 3-20. Take-Off Gross Weight

A similar trend is shown in Figure 3-21 where DOC is shown as a function of design range. The typical design convergence at long range is apparent as is the fact that the LCC design lacks the convergence because of its lack of operational sensitivity.

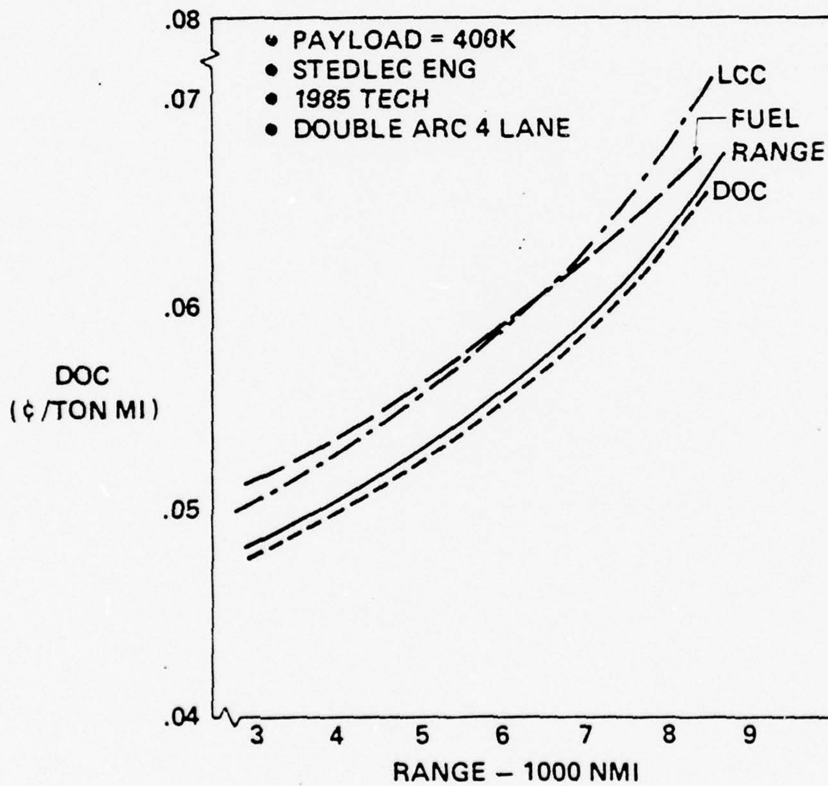


Figure 3-21. Direct Operating Costs

The mission fuel burned is shown in Figure 3-22 for the four design criteria and illustrates the asymptotic nature of the cost of achieving design ranges greater than 9000 nmi. The LCC cost design becomes increasingly inefficient as range increases. It is also interesting to note that, at the design range of 6200 nmi, the fuel used in approximately equal to the mission payload.

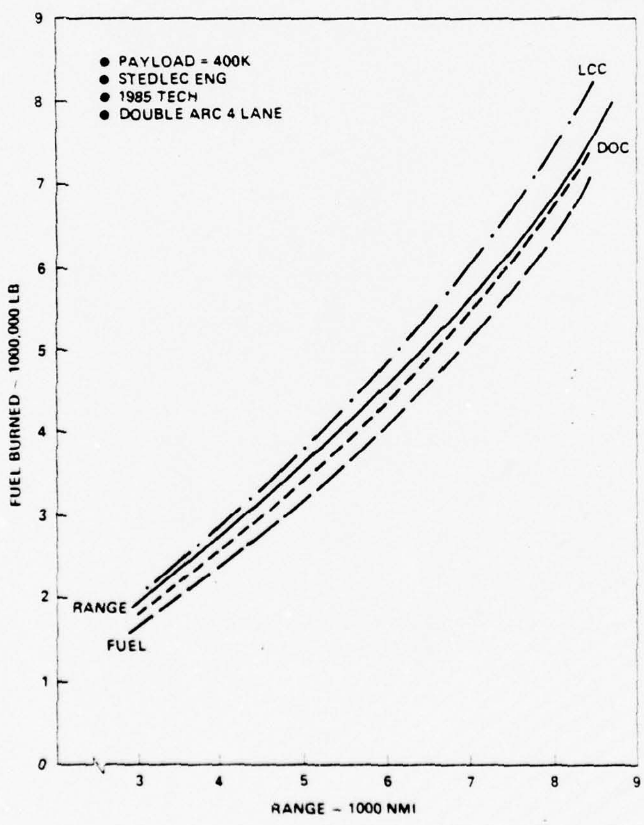


Figure 3-22. Mission Fuel Burned

3.3.3.2 Design Along The Transition Line

It is difficult in a multi-dimensional optimization process, such as has been used in this study, to access the strength of any particular optimum. Because configuration characteristics may be selected on the basis of a weak optimum, room exists for the examination of the characteristics of the design as the design criteria are varied, for example from LCC to DOC. Figure 3-23 shows the variation of LCC and DOC for designs optimized for LCC and DOC respectively. Also shown is a constant range transition line for the baseline design range. This transition line shows that a region exists in which a reduction in life cycle cost of about a billion dollars can be realized while experiencing essentially zero change in DOC. The characteristics of the designs along that transition line are shown on Figure 3-24.

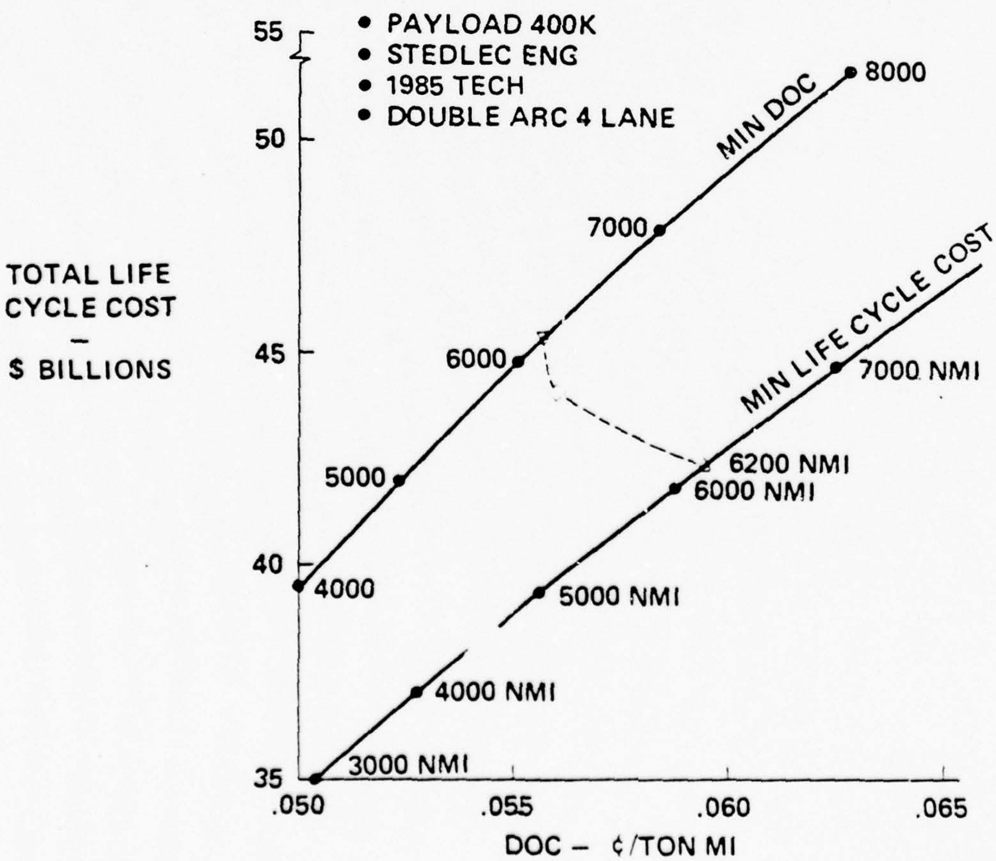


Figure 3-23. Optimum Airplane Cost Parameters Transition

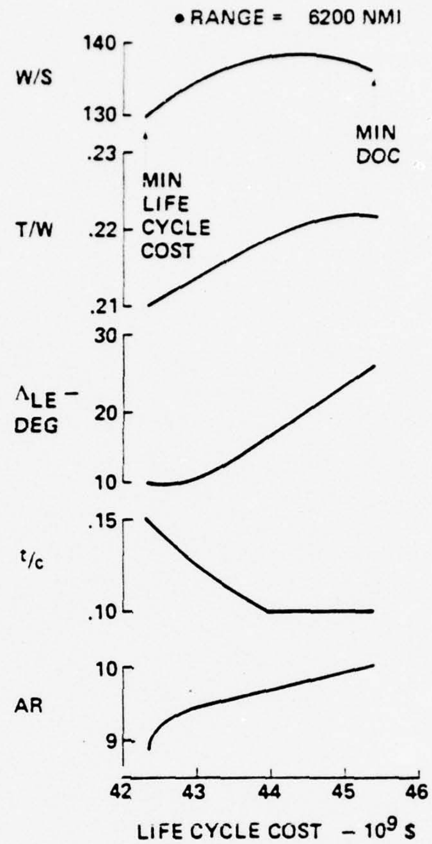


Figure 3-24. Configuration Characteristics Along the Transition Line

3.3.3.3 Design Summary

A design summary, showing the design characteristics and figures of merit for each design criteria is shown on Table 3-1, including the compromised design derived from the transition line analysis. As might be expected, the LCC design exhibited the lowest operating weight. The DOC design emphasized cruise Mach number, while the maximum range design minimized the combination of fuel and operating weight or gross weight. The minimum fuel design exhibited those tendencies for which it was designed.

The comparison between the DOC and LCC design criteria are more graphically illustrated in Figure 3-25 emphasizing the design for speed or productivity versus designing for cost.

Table 3-1. Design Options 6200 nmi Range

AIRPLANE FAMILY	MIN GWT	MIN FUEL	MIN LCC	MIN DOC	LCC/DOC COMP
CONFIGURATION DESC.					
TOGW, LBS	1460000.	1615000.	1480000.	1480000.	1461510.
W/S, LB/FT^2	148.	100.	130.	136.	139.5
T/W	.23	.184	.209	.221	.219
AWET/SREF	5.30	4.20	4.90	5.00	5.13
Λ_{LE} – DEG	15.7	22.7	10.	26.	17.
AR	9.6	12.0	8.9	10 .	9.57
t/c	.10	.10	.15	.10	.10
WEIGHTS					
OW, LB	544210.	765780.	536390.	581700.	552770.
PAYLOAD, LB	400000.	400000.	400000.	400000.	400000.
MISSION FUEL, LB	476280.	414610.	503600.	460010	470180.
RESERVES, LB	39510	34610.	40020.	38280.	38560.
PERFORMANCE					
M_{CRUISE}	.790	.780	.725	.810	.79
L/D CRUISE	24.12	30.06	23.53	24.23	24.50
SFC CRUISE	.617	.605	.596	.646	.616
RANGE FACTOR	16230.	20370.	15250.	17400.	16630.
DOC, ¢/TON NMI	5.62	5.96	5.95	5.57	5.60
LCC, BILL $	44.1	48.5	42.3	45.4	44.1

ITEM	AIRPLANE	
	DOC	LCC
Payload radius/equiv range	400,000 3600/6200	
TOGW	1,480,000	1,480,000
M cruise	.81	.72
LCC $ x 10^9	45.2 (+7.4%)	42.2
DOC ¢/ton-mi	5.55	5.91 (+6.5%)
Fuel burn	455,000	500,000 (+9.9%)
$\frac{W}{S} / \frac{T}{W}$	136/.222	130/.209
Thrust	4 @ 81,800	4 @ 76,800
Wing area	10,880	11,310
L.E. sweep	26°	10°
AR/ $\frac{t}{c}$	10/.10	8.9/.15

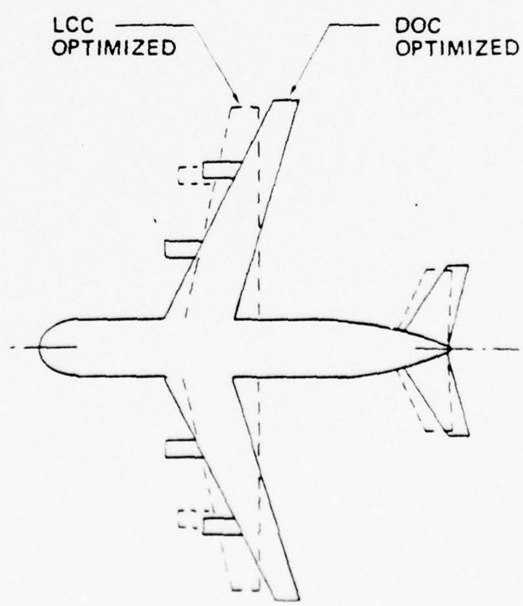


Figure 3-25. Comparison: DOC versus LCC Optimized Airplanes

3.3.4 Configuration Selection

The design experience showing the significance of design criteria on the configuration immediately placed in question the validity of life cycle cost as basis upon which to choose a configuration. It became clear that although less expensive to purchase, its performance in wartime situations, where time and hence productivity were a major factor, would penalize the effectiveness of the fleet. This conclusion was supported by the criteria implicit in the study that the use of the selected design as a commercial freighter was of prime importance. This further emphasized the importance of performance and its impact on DOC. As emphasized in Figure 3-25, an increase of 10% in fuel efficiency and 6.5% in DOC resulted from use of LCC as a design criterion compared to DOC, while an increase in 7.5% resulted from the inverse comparison.

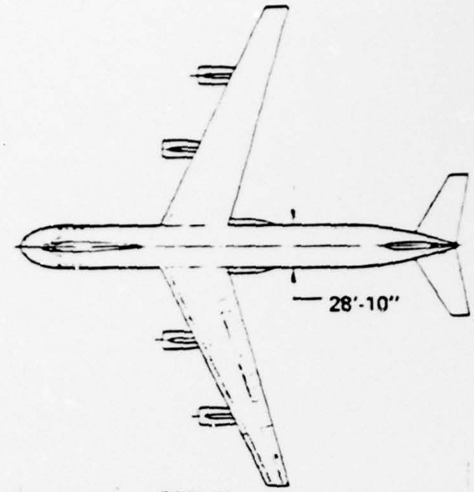
However, in commercial terms, overall profitability is highly sensitive to changes in direct operating cost. The designs which are being considered in this study have the capability to generate almost $100 billion revenue for miles of capability per year. A change of 0.1 ¢/Ton Mile could change the profit picture by 100 million dollars per year for the fleet of 250 airplanes or $400,000 per airplane per year.

Thus it was concluded that DOC would be selected as the design criteria for the baseline design because of the interest in commercial application, and because of the fallacy which apparently exists in LCC as a design criteria. This design criterion, DOC, will be representative of other possible design criteria, such as productivity, in which speed and block time play an important part.

After the design criterion was selected, further consideration was given to the problem of fuselage cross section and the implications on deck height, life cycle cost and overall performance. Another important ingredient which was considered was military flexibility as characterized by cross sectional area and floor width. The three body configurations which were considered are those shown in Figure 3-2.

Three view drawings of the three and four lane versions of the minimum DOC design are shown on Figures 3-26 and 3-27 respectively. The main differences in the configurations stem from the different fuselage length required to achieve equal floor area and pay load density. This in turn changes the height of the cargo floor. Thus the floor height of the 3 lane configuration is 14' 0" and the height of the four lane configuration is 10' 7". No kneeling has been included in the concept weights. However an additional three to five feet could be achieved in reducing deck height if kneeling were included.

RADIUS/EQUIV. RANGE		3,600/6,200	NM
TOGW		1,480,000	
PAYLOAD		400,000	LB
CRUISE SPEED	M .81 @	35,000	FT
WING AREA		10,880	SQ FT
SWEEP, L.E.		26°	
AR		10	
ENGINES	4 @	81,000	SLST
TECHNOLOGY LEVEL		1985	



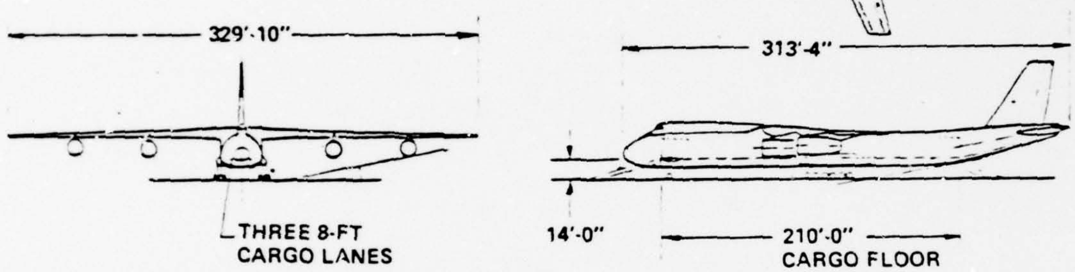
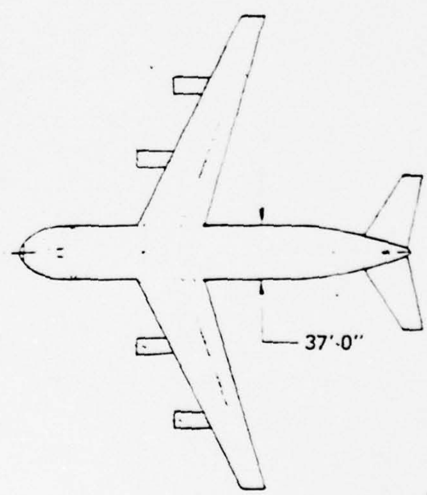


Figure 3.26. Model 1044-014 IADS Airplane Three Cargo Lanes

RADIUS/EQUIV. RANGE		3,600/6,200	NM
TOGW		1,480,000	LB
OW		581,700	LB
PAYLOAD		400,000	LB
CRUISE SPEED	M .81 @	35,000	FT
WING AREA		10,880	SQ FT
SWEEP, L.E.		26°	
AR		10	
ENGINES	4 @	81,800	SLST
TECHNOLOGY LEVEL		1985	



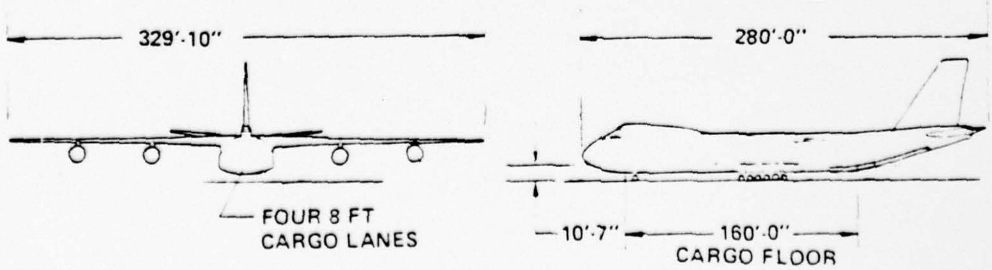


Figure 3.27. Model 1044-013 IADS Baseline Airplane Phase II – Four Cargo Lanes

The mission and design requirements are shown on Figure 3-28 and Figure 3-29.

Airplane type	Long range transport. Military and commercial versions of one basic design
Payload	400,000 lb
Military	Vehicles and cargo; outsize capability.
Commercial	Containerized; 8 x 8 ft containers
Radius (Military)	3600 nmi w/design payload. Land, unload. Takeoff w/o refuelling and return w/zero payload
Equivalent range	6200 nmi w/ design payload
T.O. field length	8000 ft
Engines	Scaled GE STEDLEC; four on wing
Technology level	1985. Airplane certification 1990

Figure 3.28. Mission Requirements IADS Reference Airplane

Design optimization basis	Minimum doc
TOGW	1,470,000 lb
W/S/T/W	136/.222
Body cross section	Four lane double arc
Cargo floor length	160 ft
Cargo floor loading (Military)	75 lb/sq ft
Payload density (Commercial)	9.75 lb/cu ft incl containers
Payload provisions	
Military	Comparable to C-5A.
Commercial	Mechanized container loading/restraint.
Passengers	No permanent provisions.
Pressurization	18,000 ft cabin altitude
Flotation characteristics, landing gear design	Similar to 747

Figure 3.29. Design Requirements IADS Reference Airplane

To give some appreciation of the size of the baseline design a more detailed cross section comparison is shown on Figure 3-30 with two M-60 main battle tanks side-by-side. The full pay load would be capable of four tanks or 32 8 x 8 x 20 ft containers. The net pay load density is 9.75 lbs/ft^2.

Figure 3-31, 3-32, and 3-33 show a graphic comparison of the baseline to the C-5A and 747 designs. As shown in Figure 3-31 the Baseline -013 has the same capability of drive through as does the C-5A.

It is worth noting that the overall length of the -013 is only 13% longer than the C-5A. However the wing span is 50% longer and the body width 60% wider.

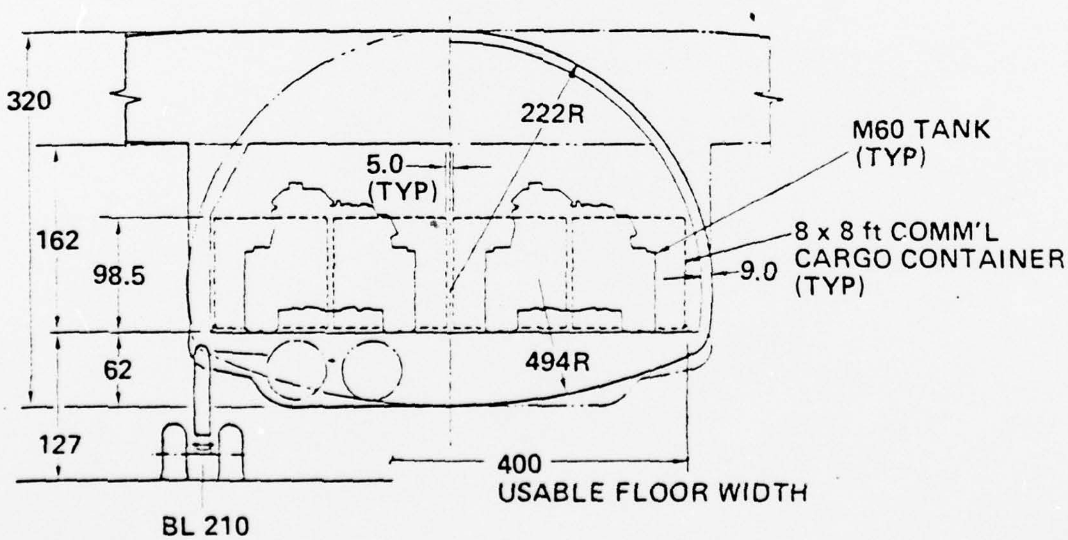


Figure 3-30. Body Cross-Section

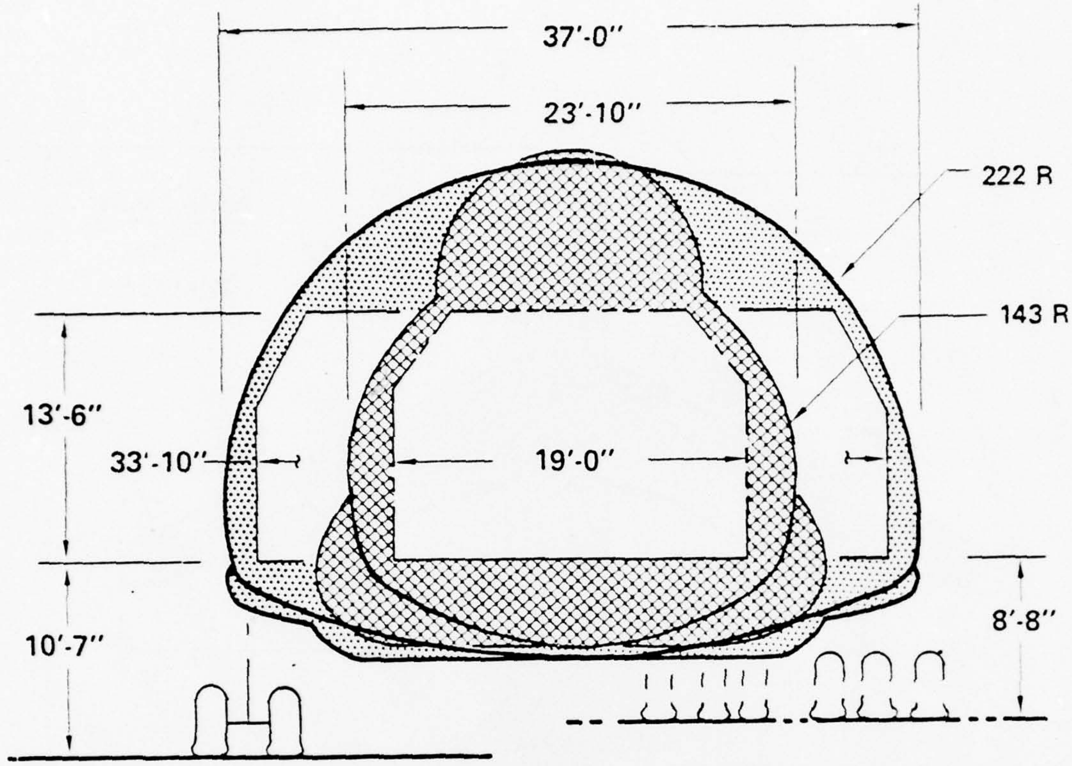


Figure 3-31. Body Cross Section

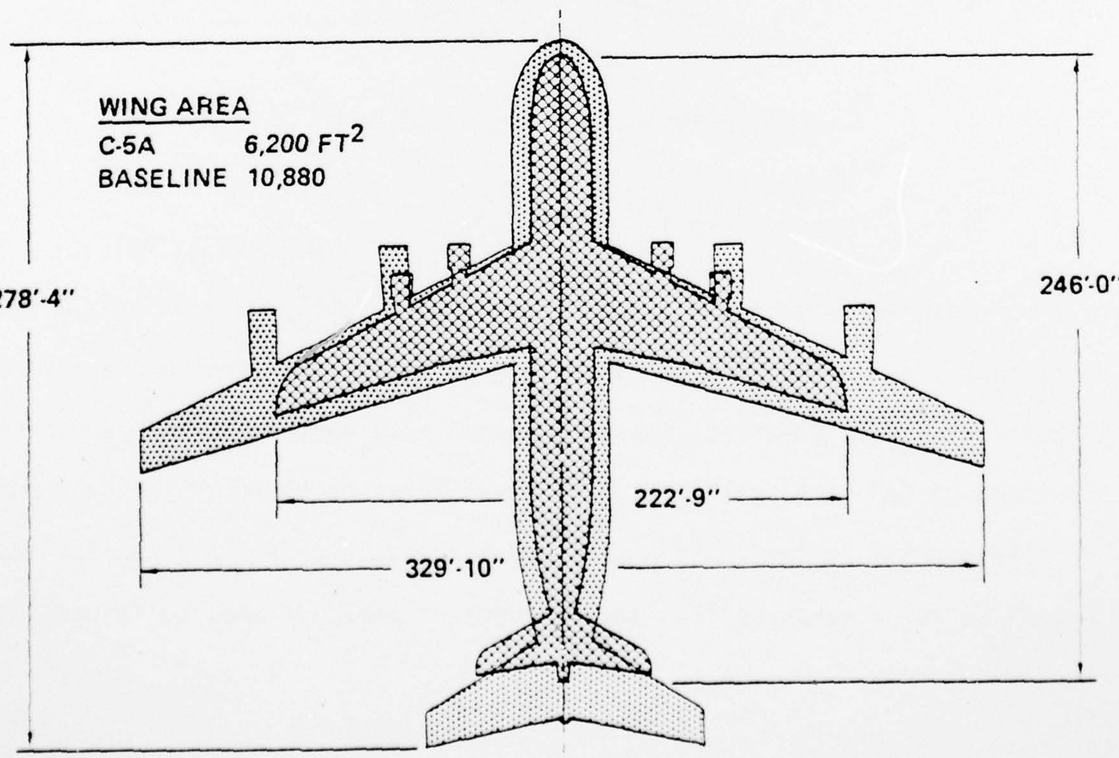


Figure 3-32. Size Comparison – C5A

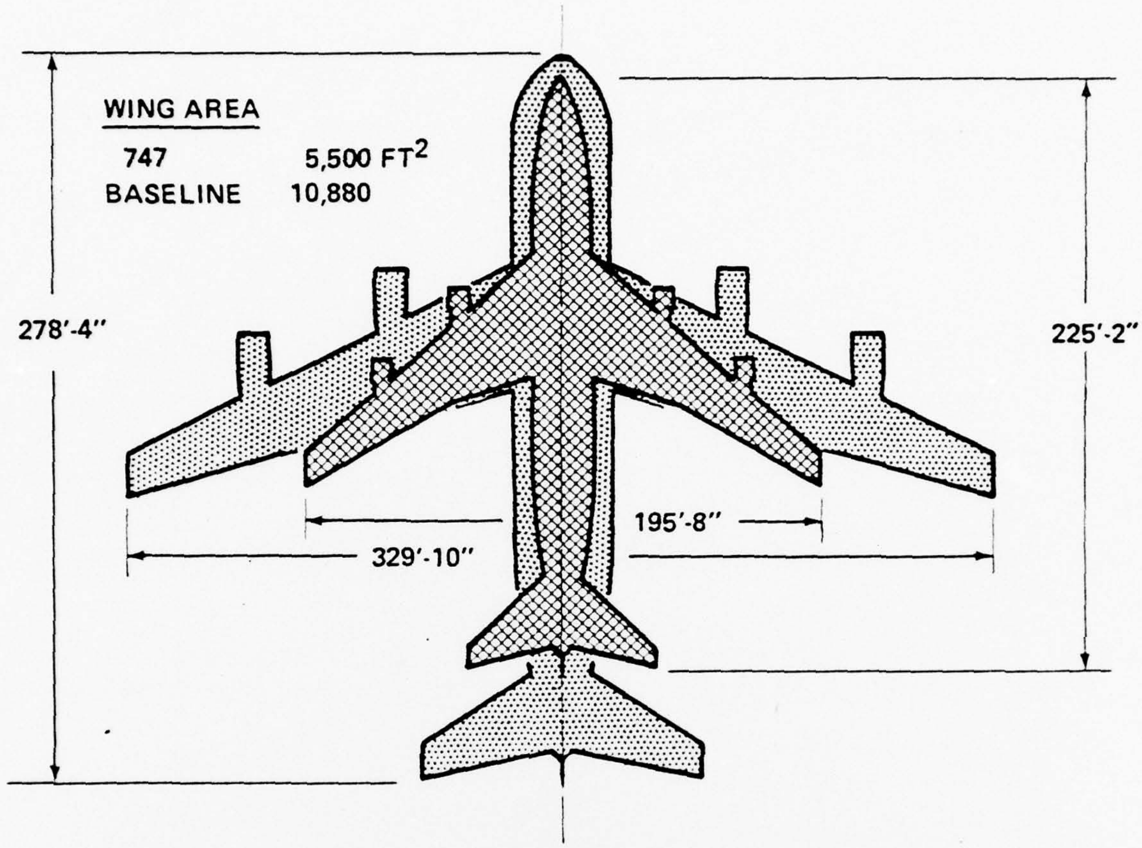


Figure 3-33. Size Comparison – 747

3.4 DESIGN VALIDATION

The design philosophy followed started with a broad based parametric design analysis and inserted increasingly greater depth of design as the mission criteria, and technology base became more well defined. The definition of the Baseline utilized analyses based on data generated by CAD to examine the influence of various design criteria and decisions on the overall configuration, and resulted in the baseline configuration shown on Figure 3-27.

The Design Validation phase provided for the detailed analysis of critical elements in the design of a large military transport to substantiate performance and costs.

Because a high level of concurrency between the technology evaluation, performance analysis, and other aspects of the study, the following approach was followed:

1) Define the Baseline configuration
2) Perform the technology evaluation and all other analysis on the Baseline, including costs.
3) Validate the Baseline by detailed analysis.
4) Show the impact of the validation on the baseline weight, and estimate the change in the gross weight due to validation analysis.

Therefore, the performance and other analyses do not reflect the validated weights but the weight initially identified in the baseline. [1]

3.4.1 Fuselage

A structural analysis of the fuselage cross-section showed that the initial radius of the lower lobe of the four lane double arc fuselage had a radius which produced an inefficient structure at 18000' cabin altitudes, and that as the cabin altitude was reduced, the structure became increasingly heavy.

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1 The validated weights agree with the initial weights to within ± 5.0% of gross weight.

The large lower lobe radius was initially selected to maintain the maximum distance possible between the keel and the ground given a fixed landing gear length, thus maximizing the kneeling capability if such a capability were desired. Increasing the radius brings the keel closer to the ground.

As a result of structural analysis, the lower lobe radius was decreased from 494.0 inches to 313.0 inches. This caused an increase in lower lobe chord depth from 62.0 inches to 100.0 inches. By this action the distance available by kneeling was reduced by 38.0" to about 20.0". Thus the floor height which could be achieved with kneeling would be about 107.0 inches or about 9 feet. The additional three feet caused a weight increment of approximately 25,000 lbs. Figure 3-34 illustrates the fuselage design solution which was followed and summarizes the impact of cabin pressure.

Lower lobe	Fuselage weight 4.5 psi	Fuselage weight 7.5 psi
Baseline	209,400	225,900
Updated	184,900	202,800

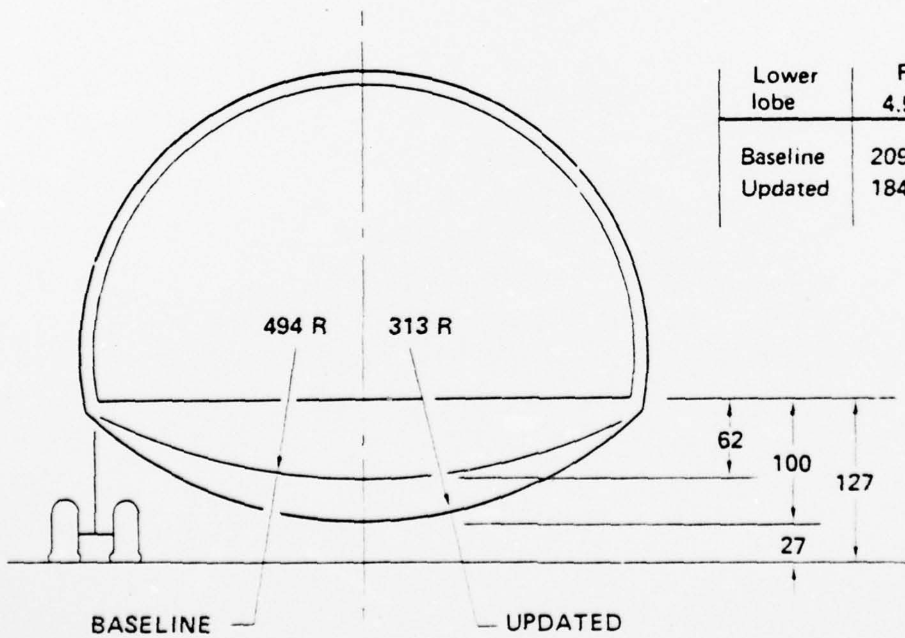

Figure 3-34. Baseline Fuselage Validation

3.4.2 Landing Gear

The analysis of the Baseline landing gear indicated that the objective to achieve the runway stress level and LCN's of the 747 was not being met.

Two solutions were proposed to achieve the desired stress level of 430 psi:

1) Increase the number of trucks from 6 to 8
2) Increase the tire size from 52 x 20.5 to 57 x 24 and increase the offset distance.

Figure 3-35 shows the variation of pavement stress as a function of the tire offset distance, and also shows the 747 design point for four main gear trucks. Also shown are tire separation distances. Landing configuration gear studies, discussed in Section 3.3.1 had shown that 6 post gears were preferable to 8 post gear for this payload size because of floor height as well as weight. Therefore, the second solution was selected as the most efficient. Increasing the tire size to 57 x 24 and increasing the separation distance, (A x B), to 55 x 74 provided the load distribution needed and achieved the desired stress level of 430 psi.

The Load Capability Number, (LCN) which is an index of the durability of the runway, is 98 for the 747 as shown in Figure 3-36. Also shown is a comparison between the Dual-Twin landing gear configuration and the four wheel truck. For single wheel load as large as must be considered here, the four wheel truck, at the offset selected for pavement stress, and at tire pressures of about 155 psi, produces an LCN equal to the 747.

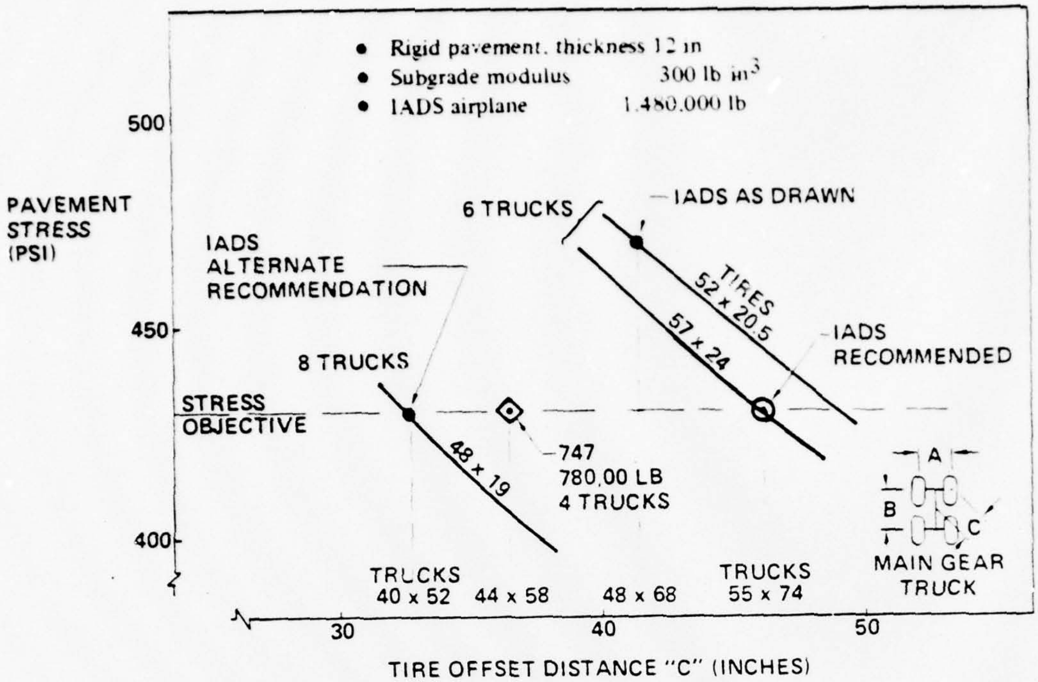

Figure 3-35. Baseline Landing Gear Validation

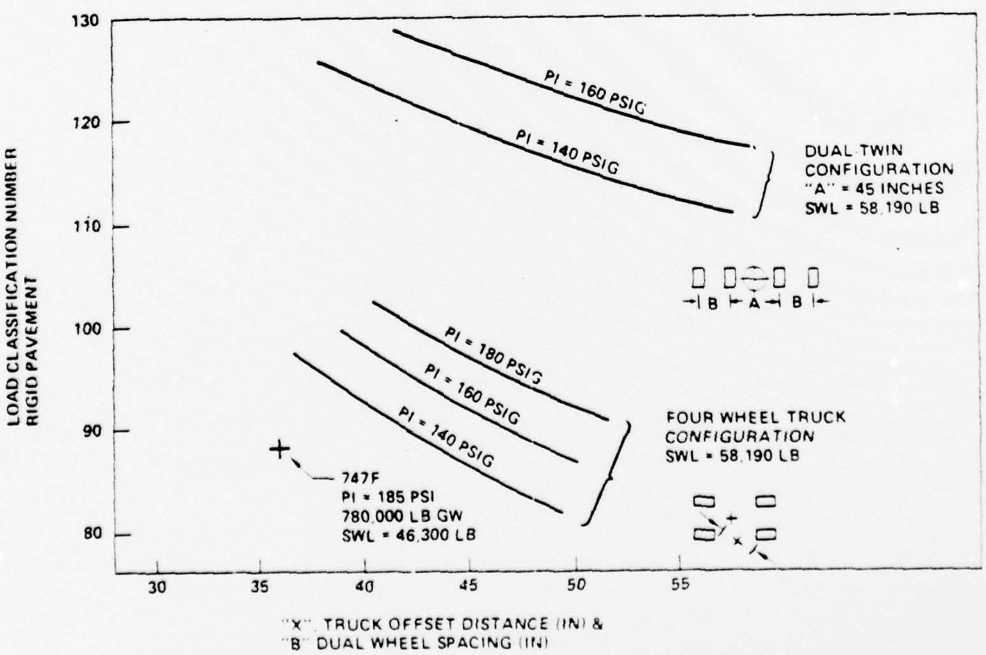

Figure 3-36. Landing Gear Analysis – Dual-Twin and Truck Gear Configuration

3.4.3 Wing

The most important assessment of the design validation process was that of providing a detailed structural analysis of the wing to confirm the strength and gust design criticality, and to define the mass properties of the wing in sufficient detail to perform a flutter analysis.

The detailed layout of the wing is shown in Figure 3-37. This wing is that which was defined in the baseline configuration, Figure 3-28. Also shown is the thickness distribution which was used on the initial design. The wing of the Innovative Aircraft Model 1044-013 was subjected to a structural analysis to establish the weight of the primary structure. The wing was first sized for strength using the computer program ORACLE. A flutter analysis was then performed to establish the flutter boundary. It was found that the configuration as drawn was deficient in flutter margin. This was corrected by a small repositioning of the outboard engine.

WING DATA

S	10,880 FT^2	TAPER	.323
SPAN	329'-10"	t/c AVE	.10
SWEEP	26°	MAC	430.7 IN
AR	10	AIRFOIL	TR 53

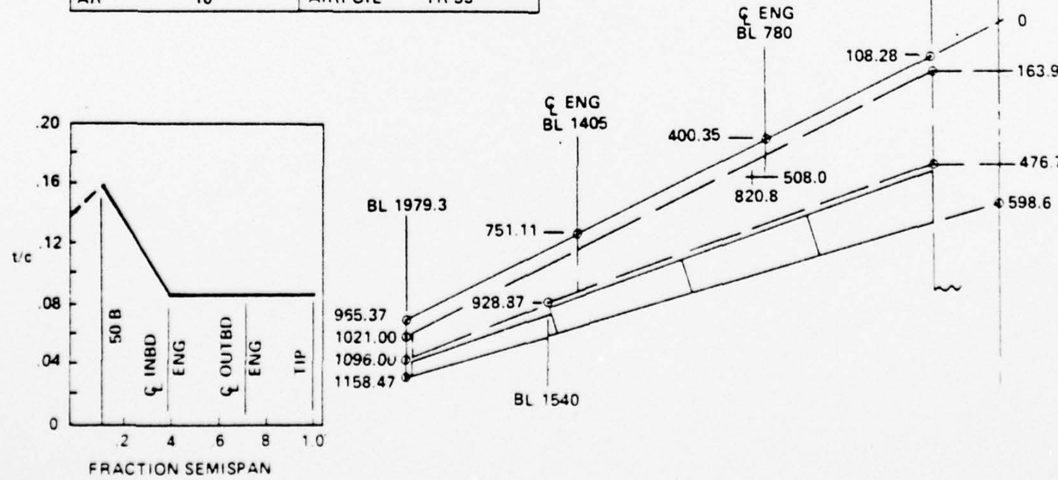

Figure 3-37. Baseline Wing Definition Structural Analysis

3.4.3.1 Structural Design Criteria

Structural Design Criteria were defined to a depth sufficient for preliminary structural analysis. They are summarized in Figure 3-38. Flight limitations were derived from the planned operational usage. Design stresses were selected such that fatigue and damage tolerance requirements of MIL-STD-1530 were met. A maximum gross weight equal to the mission gross weight was used so that payload was reduced for increased range. A 60 knot speed margin for inadvertent upsets was provided. This could be reduced if full-time speed-hold were specified for the control system. The discrete gust criteria of MIL-A-8861A was used since the design usage is not yet defined for the mission analysis criteria. So far as is known the proposed mission should not result in higher gust requirements. This aspect of strategic airlift should be investigated in case design gust velocity and fatigue considerations have a more significant impact on structure weight than has been assumed.

- Meets MIL-A-008860A series of specifications
- Basic flight design weight equals maximum take-off weight
- Maximum take-off weight = 1,480,000 lb
- Maneuver load factors are +2.5 to −1.0 (limit) at basic flight design weight
- Structural design speeds:

  V_H/M_H = 300 keas/.81

  V_L/M_L = 360 keas/.91
- Design gust velocity at V_H/M_H

  U_{d_e} = 50 ft/sec (eas) up to 20,000 ft

  U_{d_e} = 43 ft/sec (eas) at 28,800 ft

Figure 3-38. Structural Design Criteria Summary

3.4.3.2 Survey of Design Conditions

The airframe must ultimately be strong enough for all flight conditions which can possibly occur within the flight envelope. Past experience of large transport aircraft was used to isolate the most probable conditions which have a significant impact on structural design.

Figure 3-39 shows that maneuver conditions were investigated at the lowest equivalent airspeed, where aeroelastic relief is least; and at limit airspeed, where unusual effects due to the highly cambered wing section may provide critical loads. The most critical mass distributions were: (a) maximum combination of payload and fuel and (b) maximum payload and fuel associated with the minimum flight weight. Figure 3-39 also shows a taxi condition that is analyzed with an incremental c.g. acceleration of .67 g due to runway roughness. Experience with large aircraft indicates that the 1 g increment suggested in MIL-A-8862A only occurs with single axle main gears. When multiple or truck-type gears are used the incremental loading is reduced. The discrete gust conditions at V_H are also shown in Figure 3-39. The trade between the maximum gust velocity and the reduced gust velocity combined with a higher lift curve slope at higher Mach number is indicated. The complexity of the situation may be illustrated by the fact that the 43 ft/sec gust is critical without considering aeroelasticity but when aeroelasticity is included, gust loads are less than maneuver loads and the 50 ft/sec gust results in greater loads than the 43 ft/sec gust. The critical conditions for wing design indicated in Figure 3-38 do not include the effect of wing aeroelasticity.

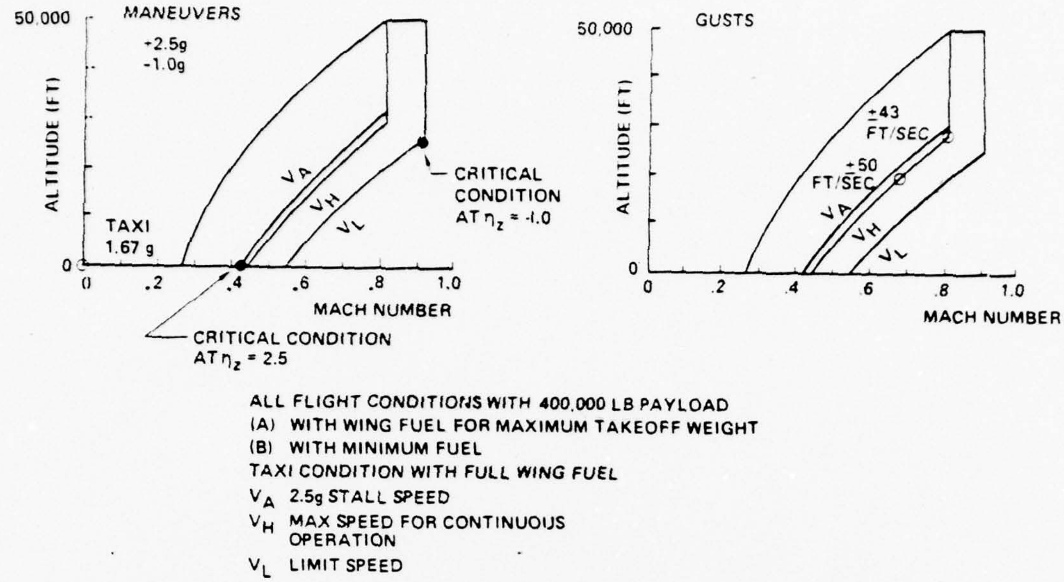

Figure 3.39. Design Condition Survey

3.4.3.3 Wing Structural Design Program (ORACLE)

Experience with large swept wing aircraft has shown that aeroelasticity must be included during the conceptual design phase of a new airplane. A multi-discipline series of design modules has been incorporated into the ORACLE system. The system sizes a fully stressed wing structure such that the aeroelastic design loads are compatible with the actual stiffness of the wing box. The information flow is shown in Figure 3-40. Some of the more significant features of the system are as follows:

1. The only inputs required are the geometry, material properties, and design conditions.
2. Dead weight loading and aerodynamic data may be input or computed internally..

3. Initial (jig) twist can be accounted for if a cruise span loading is specified.
4. Initial stiffness usually reflects a rigid structure. Convergence to a fully stressed aeroelastic solution usually occurs in 5 cycles.
5. *The flutter check is performed by a separate program.*
6. Secondary and non-optimum structure weight can be predicted when suitable data on control surfaces is available.

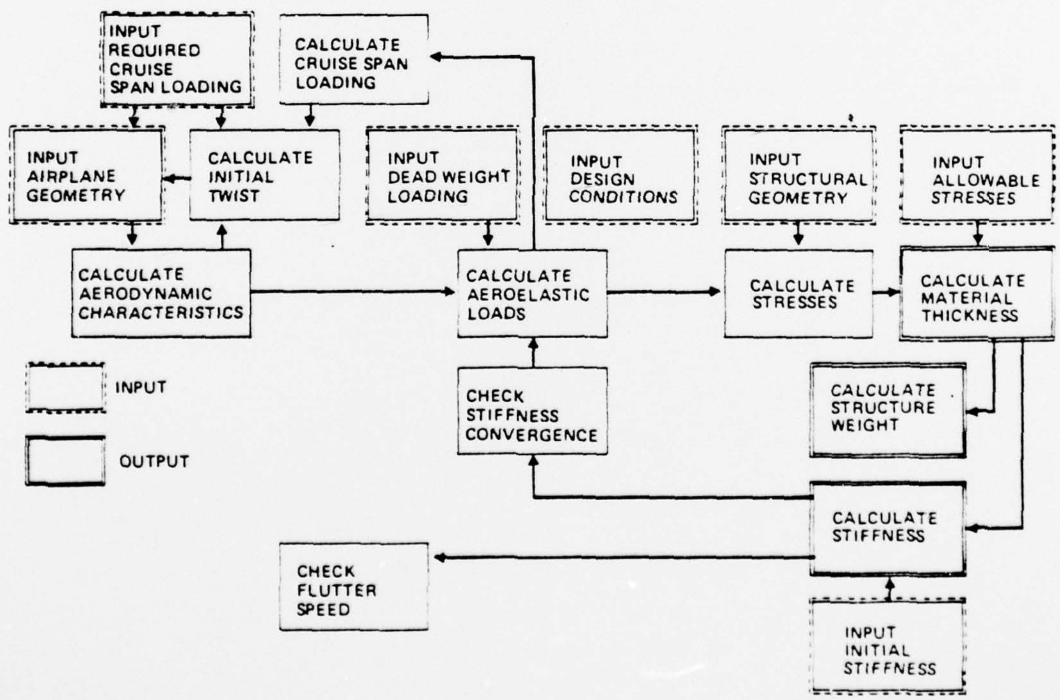

Figure 3-40. Wing Structural Design Program

3.4.3.4 Design Ultimate Wing Loads

The majority of the wing structure is critical under a 2.5 g maneuver at stall speed and maximum gross weight. Beam-type shear, bending moment, and torsion are shown in Figure 3-41. These loads include the effect of improved material properties on wing stiffness but do not reflect the use of active controls for both design limit and fatigue manuever and gust load alleviation. The critical design loads are selected from a load survey which covered 22 possible flight and ground conditions.

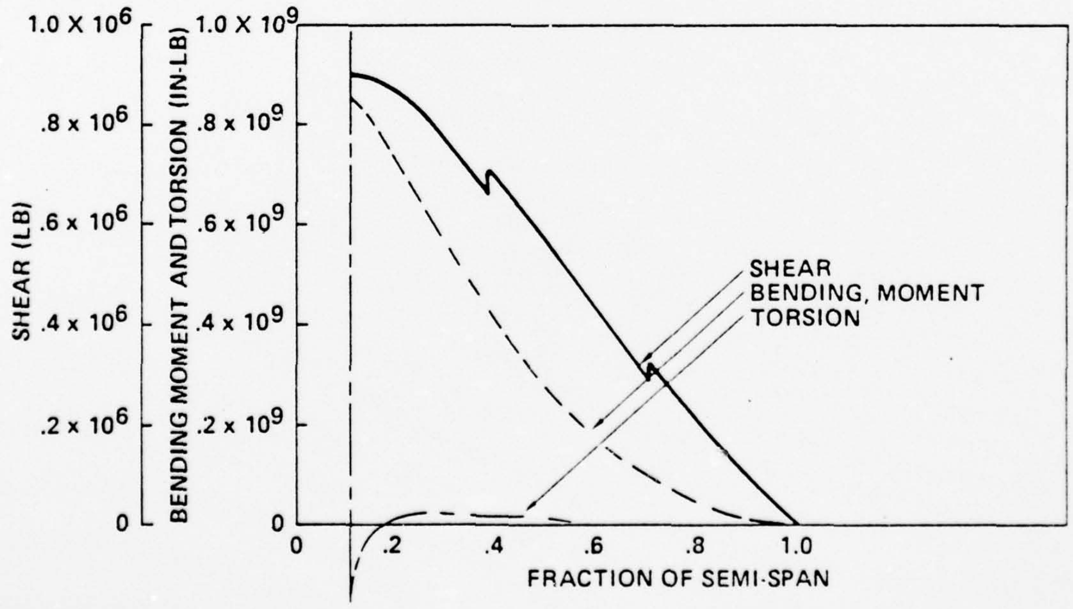

Figure 3-41. Design Ultimate Wingloads

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INNOVATIVE AIRCRAFT DESIGN STUDY (IADS), TASK II. VOLUME I.(U)
JUN 77 E A BARBER, D G BLATTNER, R C SUTTON F33615-76-C-0122

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3.4.3.5 Allowable Stresses for Wing Design

The allowable stresses used in ORACLE are presented in Figure 3-42. They represent a 5% improvement over those used for the Boeing 747. As such they reflect an upper surface of 7075 and a lower surface of 2024 aluminum alloys. They also include the differences in detail design of tension critical and compression critical structure, and construction techniques used to improve damage tolerance.

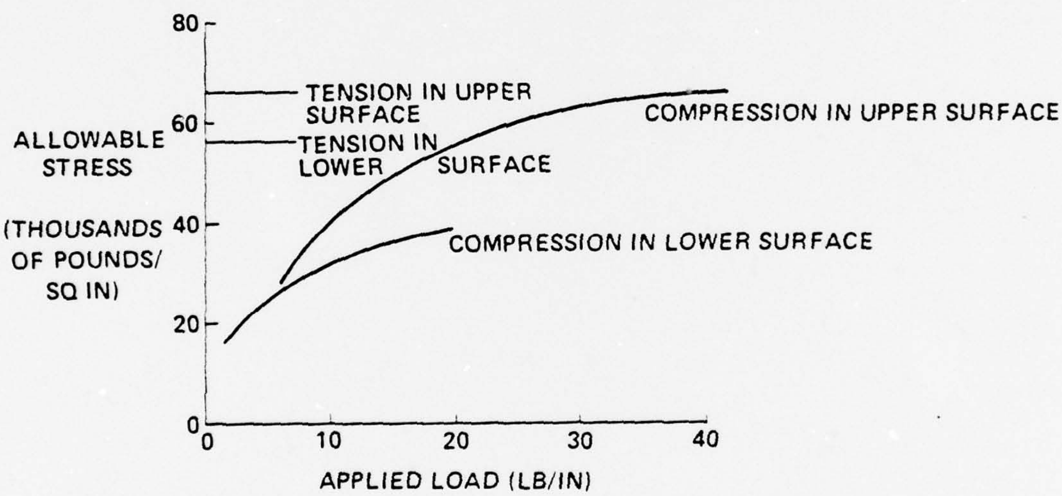
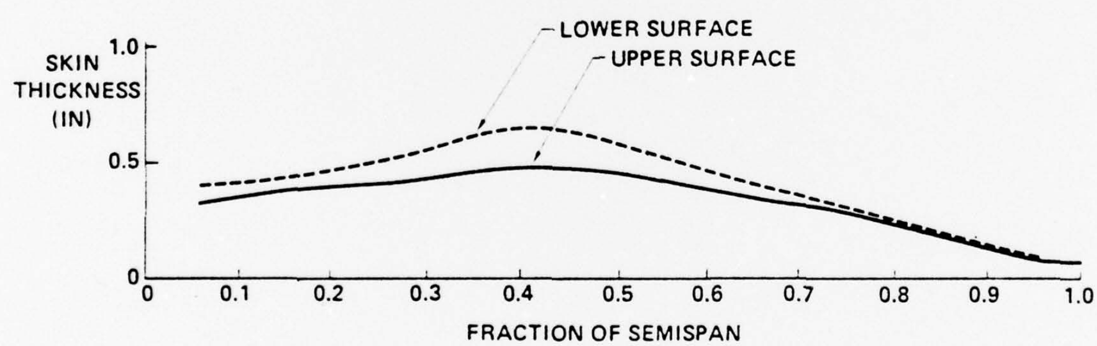


Figure 3-42. Allowable Stresses for Wing Design

3.4.3.6 Wing Stress Sizing

Wing skin and spar web/thickness resulting in a fully stressed design are shown in Figure 3-43. This data is used to compute the weight of structurally effective material. These results indicate skin thicknesses in excess of .6 inches which are deemed excessive from a damage tolerance standpoint. This indicates that further study is necessary before an actual structure for a very large airplane can be designed. The present wing configuration uses a thickness distribution which minimizes structure weight based on existing airplane sizes. It may be necessary to increase wing thickness over the outboard panels to avoid excessively thick skin panels if these cannot be reduced by adding more stiffening. Reduction in skin thickness would reduce torsional stiffness which could create aeroelastic problems. It is recommended that this aspect be studied in greater detail.



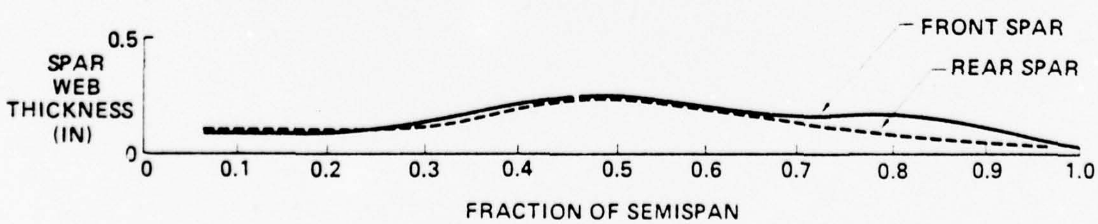


Figure 3-43. Baseline Wing Stress Sizing

3.4.3.7 Comparison of Structurally Sized and Class I Weights

Past experience is used to predict the weight of non-optimum structure and wing ribs. To this is added the weight of leading and trailing edge structure and control surfaces. This results in the wing weight shown in Figure 3-44. As noted above, the loads used to size the structure do not reflect design and fatigue load reductions available from active controls. The Class I wing weight also includes a reduction for improved analysis methods above those used for current technology aircraft. Exercises to calibrate the ORACLE prediction capability indicate that the wing weight will increase about 10 percent between the conceptual phase and fabrication. This is due to a variety of circumstances which are individually unpredictable. They include such effects as new load conditions which result from actual wind tunnel data.

Oracle prediction	Weight (lb)
Effective bending and shear material (stress sizing)	135,236
Non-optimum weight and ribs	37,491
Total box weight	172,727
Secondary structure	49,790
Basic oracle weight	222,517
Effect of load alleviation (-5.4%)	- 9,327
Effect of fatigue improvement (-2.7%)	- 4,670
Effect of improved analysis methods (– 1.8%)	- 3,109
Increase between P.D. and final weight (+10%)	+17,273
Corrected oracle weight	222,684
Class I estimate	220,360

Figure 3-44. Baseline Wing Weight Validation

3.4.3.8 Flutter Analysis

A flutter analysis of the Innovative Aircraft model 1044-013-2 wing shows a flutter speed which is below the required flutter speed. The wing flutter speed for the critical condition, full fuel, is .71 V req. For zero fuel the flutter speed is 1.12 V req. Trade studies were run on the outboard nacelle location which indicated that adequate flutter clearance could be obtained by moving the outboard nacelle from 71% to 73% of span. Moving the outboard nacelle aft from its baseline location had little effect on flutter speed, Figure 3-45.

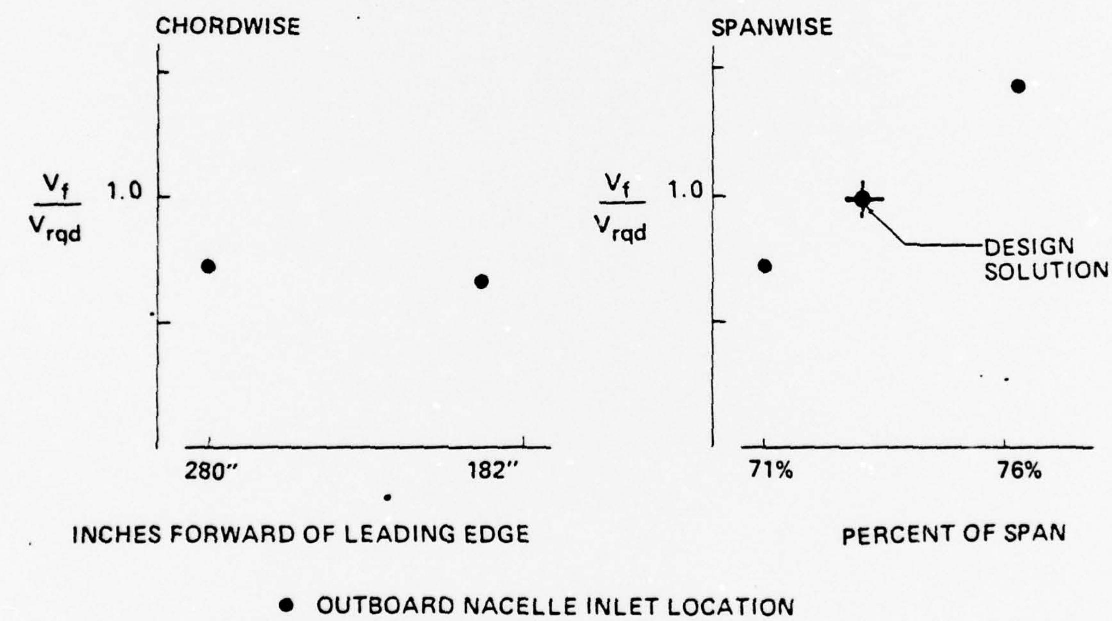


Figure 3-45. Effect of Outboard Nacelle Location on Flutter Speed

3.4.4 Design Validation Summary

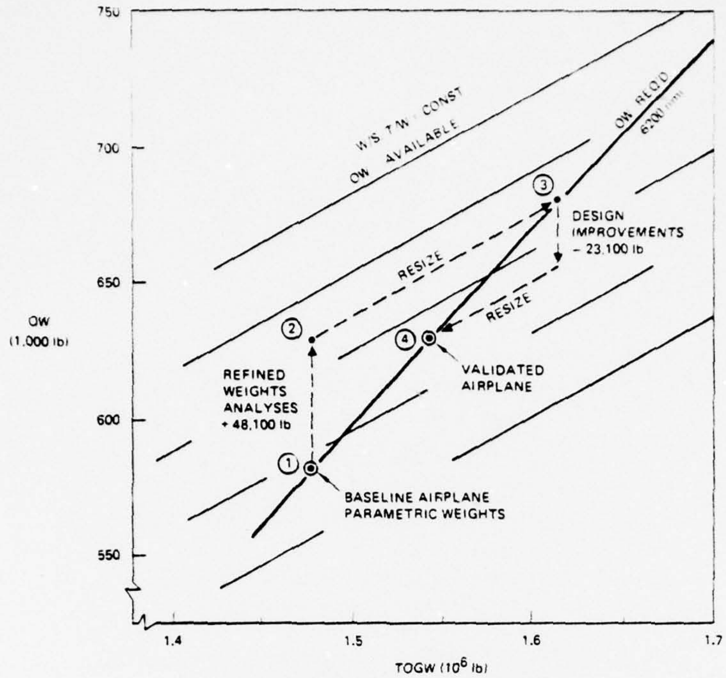
The design validation process indicated the need for configuration changes. The body cross section was drawn initially with a lower lobe radius too large to provide an efficient structure at lower cabin altitudes. Increasing the curvature of the lower lobe, although compromising the potential for a kneeled deck height, produced a fuselage structure which was approximately 25,000 lb. lighter.

The landing gear arrangement, in order to achieve pavement stress levels and LCNs comparable to the 747, required additional off sets and larger tires. This increased the landing gear system weight by 1400 lb.

The detailed weights buildup resulting from the Class II analysis show a difference of 48,000 lbs. compared to the initial baseline weights. This when resized to 6200 nmi, resulted in a gross weight of 1.62 million lbs. Including the design changes due to reduction of the body cross section lower lobe radius and the increase in the landing gear group weight produced a net decrease in operating weight. This, when the design was resized produced a decrease in OW of 27,000 lbs. The net result of the increase in OW due to detailed weight analysis, design changes, and resizing, resulted in a net increase in OW to 629,000 lbs. and an increase in take off gross weight to 1.54 million lbs., compared to the initial base line gross weight of 1.48 million lbs.

This validation process is illustrated on Figures 3-46 and the weight breakdown shown on Table 3-2. The group weight summary is shown on Table 3-3.

The initial OW was increased by 8.1% and the gross weight by 4.3% by the validation process. These are considered to be well within the level of accuracy of the analytical technique and engineering data used in the study.



	① BASELINE AIRPLANE	④ VALIDATED AIRPLANE	CHANGE
TOGW	1,480,000	1,544,000	+4.3%
OW	581,700	629,000	+8.1%

Figure 3-46. Validation and Final Sizing Summary

Table 3-2. Summary Data Validation and Sizing Group Weight Statement—Resized Design

• 1985 Technology

Item		Airplane				
		① Baseline configuration parametric weights	② Baseline configuration class I and II weights analyses	③ Airplane ② resized for 6,200 nmi	Design changes and resizing	④ Validated airplane (Airplane ③ with design changes + resizing for 6,200 nmi)
Range	nmi	6,200	5,460 (Fallout)	6,200	• Increase lower lobe depth (-24,500 lb) • Adjust landing gear size to meet flotation *objective* (+1400 lb) • Resize revised airplane for 6,200 nmi (-27,000 lb)	6,200
Wing area (135 psf)	ft^2	10,880	10,880	11,882		11,324
OW	lb	581,700	629,800	679,100		629,000
Payload	lb	400,000	400,000	400,000		400,000
Fuel	lb	498,300	450,200	536,900		515,000
TOGW	lb	1,480,000	1,480,000	1,616,000		1,544,000

Table 3-3. Group Weight Statement

Validated Model 1044-013	Current Technology Weights (Lb)	1985 Technology Weights (Lb)
Wing	260,000	232,700
Horizontal Tail	11,620	10,400
Vertical Tail	17,840	10,600
Body	184,530	177,700
Main Gear	60,250	58,260
Nose Gear	7,940	7,940
Nacelle or Eng Section	23,100	23,100
Air Induction		
Structure	(559,200)	(520,700)
Engine	58,820	58,820
Engine Accessories	720	720
Fuel System	4,960	4,960
Engine Controls	240	240
Starting System	160	160
Propulsion	(64,900)	(64,900)
Auxiliary Power Unit	930	930
Instruments & Nav Equip	860	860
Surface Controls	11,880	9,280
Hydraulic/Pneumatic	7,870	5,510
Electrical	4,100	4,100
Avionics	3,450	3,450
Furnishings & Equip	8,490	8,490
Air Cond & Anti-Icing	5,610	5,610
Auxiliary Gear & Tie Down Chains	1,770	1,770
Fixed Equipment	(44,960)	(40,000)
Weight Empty	669,060	625,600
Crew	1,290	1,290
Crew Provisions	180	180
Oil & Trapped Oil	500	500
Unavailable Fuel	1,400	1,430
Payload Provisions		
Weapon Bay Fuel Prov		
Non-Exp Useful Load	(3,370)	(3,400)
Operating Weight	672,430	629,000
Payload (Incl Exp Pen Aids)	400,000	400,000
Fuel-Wing	471,570	515,000
Fuel-Body		
Gross Weight	1,544,000	1,544,000

3.5.0 TECHNOLOGY ASSESSMENT

This section identifies, describes and quantifies specific advanced technology concepts applicable to the next generation military logistics aircraft. Operational considerations and developmental costs are also presented to assist in selection of those technology concepts to be incorporated in specific point designs. The paragraphs which follow describe technology advances which could be incorporated into a 1990-2000 IOC military airplane.

3.5.1 Wing Aerodynamics Design

Present day computational techniques include the effects of viscosity and allow the design of optimum sections for any given application. Figure 3-47 shows the progressive improvement in two-dimensional performance which has taken place over the past twenty years. It is presented here as an increase in critical Mach number at 10% thickness and zero sweep. This can be adapted to a particular configuration to provide increased thickness for reduced weight or reduced sweep back for improved low speed performance. Figure 3-48 shows the comparison between conventional three-dimensional wing performance and what would be available if every section of the wing were operating as its achievable two-dimensional performance.

3.5.1.1 Improvement Options

Increased fuel costs will continued to drive a trend toward higher value of wing aspect ratio. The trade between wing weight and aerodynamic efficiency must be evaluated in the economic frame predicted for the years of service. A powerful effect in reducing weight penalties for high aspect ratio will be the introduction of active controls both for flutter suppression and gust or maneuver load alleviation.

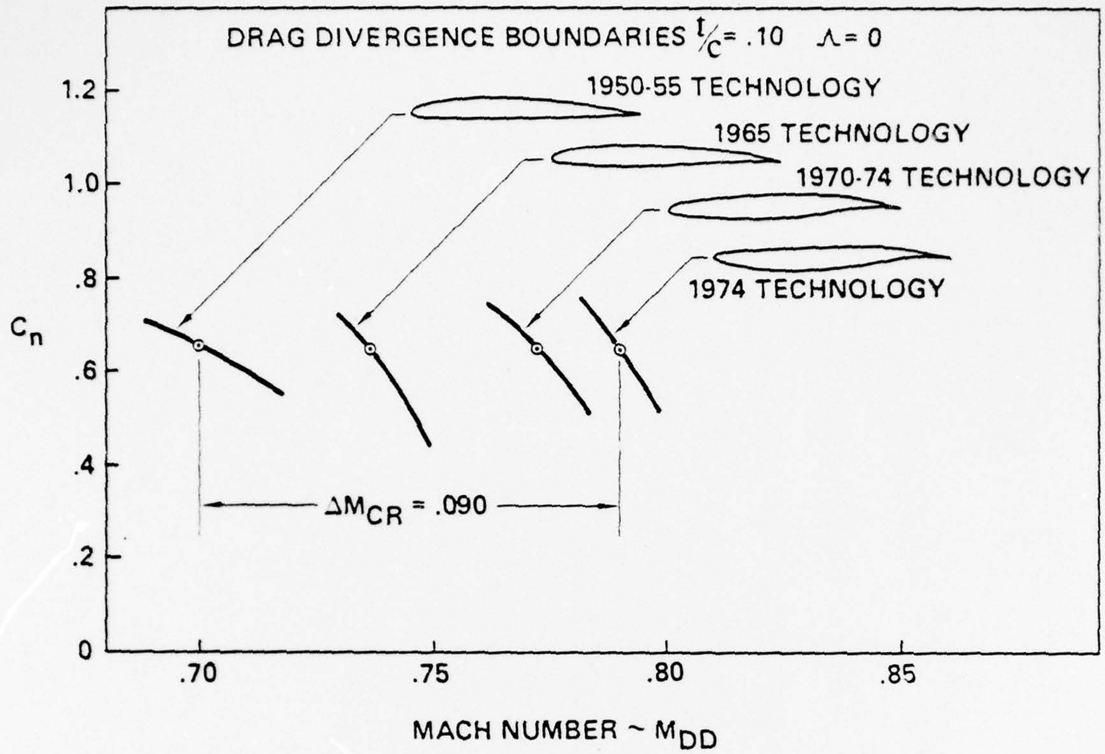


Figure 3-47. Airfoil Technology Development

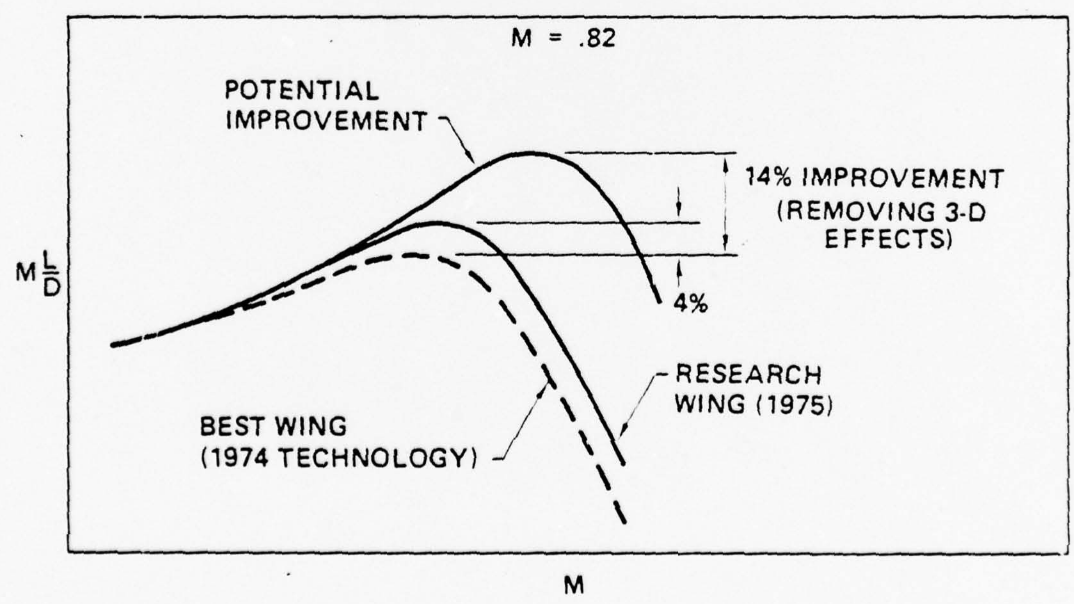


Figure 3-48. Three-Dimensional Wing Performance

Another means of obtaining high aspect ratio is the use of the braced wing. It may be possible to design the strut profile so as to maintain natural laminar flow.

Laminar flow control by means of suction is today considered feasible, but a great deal of development work is required to identify an efficient structural arrangement which incorporates the required ducting. It should be considered as an attractive possibility for the 1995-2000 IOC time frame.

3.5.1.2 Low Speed

The opportunities for improvement in low speed design are substantial and are indicated in Figure 3-49 which compares levels of lift/drag ratio currently achieved with the maximum attainable. Improvements will occur from the elimination of flap cutouts, larger flap span and increased flap efficiency from better detailed design and the addition of non-planar features such as vertical fins at the extremes of flap segments.

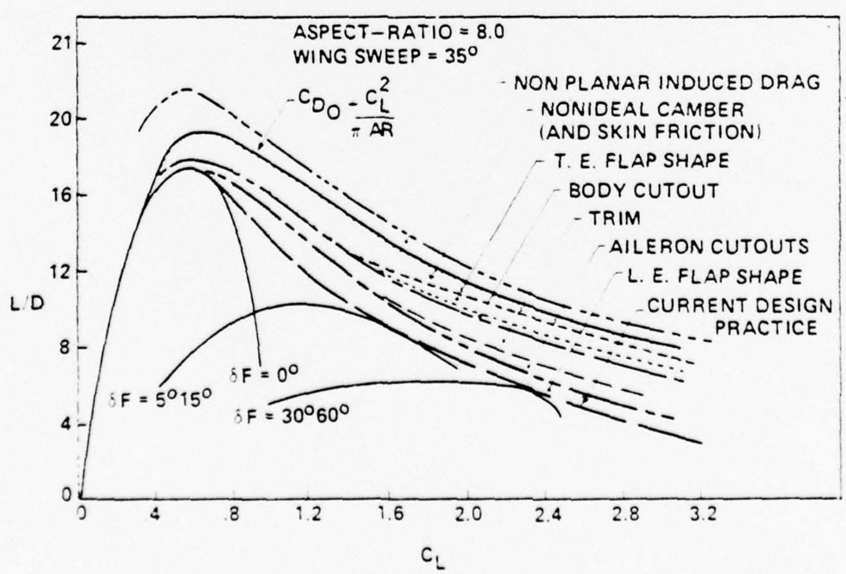


Figure 3-49. Low Speed, High-Lift Configuration

3.5.2 Empennage

The four-engined high-aspect ratio wing design requires a larger vertical tail unless automatic control systems can be relied upon to counter the effect of engine failure at takeoff. Relaxed longitudinal static stability will minimize horizontal tail size. Careful aerodynamic design will eliminate unfavorable interference and minimize trim drag. The horizontal tail must be as carefully shaped as the wing. This will entail a detailed knowledge of the downwash field of the wing and the flows induced by the boattailing of the aft body. Afterbody lines are a critical aspect of efficient military transport design due to the requirements of aft loading, airdrop, and missile launch.

3.5.3 Propulsion

3.5.3.1 Nacelle Design

Integration of the propulsion nacelle on the wing currently receives a great deal of design consideration. Spanwise nacelle positioning and strut stiffness are important parameters in flutter characteristics. In both overwing and underwing installations, the design must be carefully tailored to obtain best overall performance. Figure 3-50 shows as an example a strut-mounted over-wing configuration where the final contour lines provide a better critical Mach number than that of the wing alone.

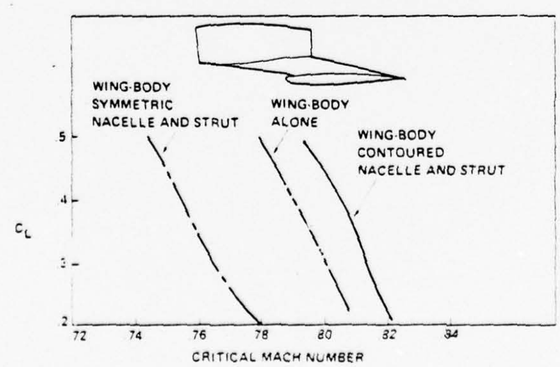


Figure 3-50. Drag Effect of Contouring Overwing Strut-Mounted Nacelles

3.5.3.2 Propulsion Technology

Improved analytical aerodynamic and mechanical design techniques will lead to significant improvements in SFC with small reductions in weight. These improvements result from improved blade tip and intra-blade row analytical design procedures, providing increased component load factors and efficiencies. Blades will be locally shaped to reduce the loads associated with part span shrouds where these are required.

Application of advanced flow field prediction procedures to the internal and external flow regimes of a nacelle installation will minimize installed SFC. The use of 3-D viscous flow procedures will enable the complex geometry of the internal exhaust mixers to be optimized to achieve maximum net thrust.

Advances in engine hot section design will improve thrust and SFC by reducing cooling flow requirements and allow increases of up to 300°F in turbine inlet temperature.

Engine life cycle cost estimation during development can be used to identify those technologies which provide additional performance at acceptable costs, to understand how the costs vary in different applications, and to assist in logistic and maintenance studies.

Appendix C is a description of the 1985 technology engines.

3.5.4 Structures Technology

The use of advanced structural materials will lead to weight reductions in all components of the airplane. High purity aluminum alloys having higher strength and toughness, are now in the laboratory stage and should be ready for use in the 1985 technology baseline. Graphite-epoxy honeycomb structure will have increasing application as more design data and experience is obtained, but production status will probably not be achieved until 1990 or later. Composite primary structure will be 15 to 25% lighter than aluminum structure.

Maneuver load control will redistribute wing lift by moving the center of lift inboard and thus reduce wing bending moments. Gust load control will reduce peak transient loads in the airplane while fatigue life will be increased by the use of active controls to reduce the amplitude and number of transient bending cycles due to continuous turbulence. Flutter mode suppression will actively damp flutter modes using the aerodynamic control surfaces. These items will all emerge from development work integrating structure, aerodynamic and flight control technologies.

Improved performance and lowered airframe weight will be obtained in the 1985 time frame through smaller control surface areas and reduced wing box weight.

More powerful stability analysis techniques will provide new data for optimization of skin panels, wing ribs and body frames. Pressurized cabin stress analysis will provide more detailed definition of the skin stress distribution. Improved finite element analysis will be aimed at better prediction of stresses at ultimate load. Damage tolerance analysis will provide data for body pressure loading and could result in reductions of up to 10% in body skin gage in pressure critical areas.

The overall reduction in structure weight which might accrue from these developments might be expected to be in the range of 15% to 20% with an upper limit of 25%.

3.5.5 Mechanical Electrical Systems

3.5.5.1 Secondary Power

The introduction of active controls for flutter suppression and gust or aerodynamic maneuver load alleviation will impose greater demands on the secondary power systems. Achieving these requirements with present day hydraulic and/or electrical power systems technology would result in large weight penalties in the power generation, distribution and actuation systems

3.5.5.2 Technology

The use of advanced high pressure hydraulic systems will reduce hydraulic distribution and actuation system weight. Further weight reduction will be achieved through improved distribution systems using higher strength tubing and fitting and improved assembly techniques.

Use of stored energy devices such as flywheels could result in further reduction in power distribution system weight. Advanced fluids with increased fire and erosion resistance will be available for improved safety and reduced life cycle costs.

Advanced variable speed, constant frequency (VSCF) permanent magnet starter-generator systems will be available for incorporation into a 1985-1995 Technology airplane. This coupled with advanced multiplexing distribution system and improved backup power technology will result in considerable electrical system weight reduction.

Laminar flow control by means of suction will directly affect the environmental control system by its impact on bleed air requirements. Further, bleed extraction capability of future high bypass ratio engines is lower than for present day engines, thus resulting in greater operating penalty from bleed air extraction.

Bleed air extraction and ECS weight will be reduced through use of advanced, on demand and centrally controlled ECS systems incorporating:

- Cabin air recirculation through improved filtration and cooling concepts
- Advanced hybrid avionics cooling concepts
- Closed loop concepts

Improved steady state and transient analysis techniques will aid in achieving significant secondary power systems weight reduction through improved load requirements prediction.

3.5.5.3 Landing Gear Systems

Advanced landing gear systems currently under study or in the development phase include:

- Limited slip skid control system and concepts for reduced tire and brake wear and improved safety of operation.
- Advanced steering system concepts for improved runway operating safety. Concepts for steering multiple main and nose gear are required for large transport aircraft.
- Adaptive landing gear concepts for dynamic load alleviation on rough runways and taxiways. Adaptation of these concepts will improve structural fatigue life and reduce structure weight.
- Light weight composite carbon brakes offer considerable weight savings and are currently under development to improve wear life and reduce production costs. The advanced skid control concepts should reduce brake and tire wear by as much as 30 percent. Brake weight is reduced by one-third with structural carbon heat sink material.

3.5.5.4 Cargo Loading Systems

Advanced mechanisms analysis for spatial linkages will aid in design of kneeling systems, ramps and large cargo door actuation and loading mechanisms for outsize cargo accommodation. Heads up display and indirect viewing will permit relocation of the cockpit, thus permitting new schemes for cargo loading systems.

3.5.6 Technology Assessment Summary

Detailed discussion of the technology assessments are provided in Appendix E. Summaries of the analysis of technology contained in Appendix E are presented in Tables 3-4 and 3-5. The columns on the right side of these figures represent those technologies selected for the 1985 baseline and those selected for evaluation as advanced technologies, available for inclusion in a program with an IOC from 1990-2000. The summary for the 1985 baseline has been presented as Figure 2-18. The dollar values shown represent assessments of dollars needed to develop the technologies, not the system development costs, or for example the engine development costs.

The specific advanced technologies identified for further evaluation as Advanced Technologies are:

1. Laminar Flow Control
2. Prop Fan
3. Advanced Composite Primary Structure.

Summing the potential system performance gains possible for the 1985 and 1995 technology airplanes is difficult because of uncertainties in development risk, development costs, and integrated design effects. However, the potential exists for substantial gains in aerodynamic, propulsion and weight technology performance for the 1985 technology and nearly double those gains are apparent for the 1995 technology - but, at technology development costs between five and ten times that of the 1985 base.

Table 3-4. Technology Assessment—1

	Probable Technology Date of Maturity 1975	1980	1985	1990	Selections for 1985 Tech Baseline	Selections for 1995 Adv system
Aerodynamic Tech:						
Variable camber		▽			✓	✓
High AR		▽				
Wing tip fins		▽				
Adv airfoils (Lo Δ M)		▽			✓	
Adv airfoils (Hi Δ M)			▽			
LFC			▽			✓
Body BLC			▽			✓
Adv aero methods	▽				✓	✓
Potential performance improvement					9% M L/D	25%-35% M L/D
R&D cost, dollars					10M-20M	120M-240M. .
Propulsion Tech:						
Adv component aerodynamics			▽		✓	✓
Adv materials			▽		✓	✓
Electronic fuel control		▽			✓	✓
Eng/airframe structure integration		▽			✓	✓
Improved Nacelle aerodynamics		▽			✓	✓
New eng (TF) development					✓	
New eng (Prop/fan) development			▽			✓
Potential perf improvement					12-16% SFC	15-20% SFC
R&D cost, dollars					500M-800M	500M-1000M

Table 3-5. Technology Assessment—2

	Probable Technology Date of Maturity 1975	1980	1985	1990	Selections for 1985 Tech Baseline	Selections for 1995 Adv System
<u>Structures Technology</u>						
Active controls		▽			✓	✓
Improved alloys		▽			✓	✓
Composite primary structure			▽			✓
Structural arrangement		▽			✓	✓
Adv structural analy methods		▽			✓	✓
Manufacturing improvements		▽			✓	✓
Potential weight improvement					7% struct	20-30% struct
R&D cost — Dollars					18M-24M	170M-320M
<u>Mech/Elec Systems Technology</u>						
Secondary power and control system mechanization		▽			✓	✓
ECS – avionics cooling		▽			✓	✓
Landing gear systems		▽			✓	✓
Potential perf/weight improvement					1- 3% SFC,	Δwt 4000 - 6000 lb
R&D cost — Dollars					14M-25M	

4.0-Summary-Advanced Technology, Design and Mission Sensitivity

Advanced technology and design innovation

- Prop fan
- Advanced composites
- Laminar flow control
- Improved low-speed aerodynamics
- Boost engines
- Strut-braced wings
- ACLS

Mission sensitivity

- Payload and range
- Endurance
- Field length

4.0.0 ADVANCED TECHNOLOGY, DESIGN AND MISSION SENSITIVITY

The definition of the baseline configuration provided a basis upon which innovative designs, advanced technology, and sensitivity of the baseline mission to changes in mission criteria could be examined. Because of concurrence in the study effort the initial baseline was used as a reference rather than the validated design.

4.1.0 ADVANCED TECHNOLOGY AND DESIGN

The approach used the baseline configuration and planform to evaluate the innovative designs rather than reconfiguring for each design. However, exceptions were made in the case of the propfan, where a low Mach number (LCC) configuration as well as a high Mach number (DOC) configuration were considered; and the LFC configuration where lower wing loadings were advisable in order to fairly evaluate the technology and design. Also, a higher aspect ratio wing was used in the case of the strut braced wing.

4.1.1 Prop Fan

Considerable effort has gone into the exploratory design of a new family of turbo prop engines which might provide a significantly improved energy efficiency over that of high by pass ratio turbofans, References 5 and 6. An evaluation was made of a prop fan variation of the baseline configuration as shown in Figure 4-1. The improved capability of the propfan operating at M=0.80 was a 6% increase in range over the turbofan engine. A heavier operating weight was caused primarily by the heavier engine/gear box/propeller installation of the prop fan. An 11% reduction in SFC was used in calculating the performance of the propfan configuration, Figure C-1, Appendix C.

The impact of propfan propulsion was also considered for endurance missions where a lower cruise Mach number might show the propfan to better advantage. A propfan configuration based on a design optimized to minimize life cycle cost was used to illustrate the significance of cruise Mach number because of its lower cruise Mach number. This configuration is shown in Figure 4-1 with turbo fan propulsion. Figure 4-2 shows the variation of endurance with radius of action for both the low speed LCC and higher speed DOC designs. The radius at zero time on station is improved from 3100 to 3200 nautical miles[(1)] for $M_{cruise} = 0.80$ and from 3100 to 3400 nautical miles at $M_{cruise} = 0.70$. The major impact of propfan technology, however, occurs for station keeping missions. The propfan can improve zero radius endurance from 15 hours to 22 hours, or 45%, over a DOC configured turbofan, or slightly less than that when compared to a LCC configured turbofan.

4.1.2 Advanced Composite Primary Structure

Considerable research has been devoted to the use of boron and graphite filaments in lightweight composite structure for aircraft. The NASA has made advanced composite structures a keystone of its Aircraft Energy Efficiency program (ACEE), and its flight demonstration program for composite primary is scheduled for completion by the mid 1980's. Secondary structure and empennage components are already being flight evaluated. The purpose of this evaluation was to illustrate the improved capability which might be possible from the use of advanced composite structures for an Advanced Military Transport.

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(1) Payload is 400,000 lbs in and out.

RADIUS/EQUIVALENT RANGE	3830/6600 NM
TAKEOFF GROSS WEIGHT	1,480,000 LB
OPERATING WEIGHT	603,900 LB
PAYLOAD	400,000 LB
CRUISE SPEED	M.80 @ 30,000 FT
WING AREA	10,880 SQ FT
SWEEP, L.E.	26°
AR	10
ENGINES	4 @ 55,000 SHP
TECHNOLOGY LEVEL	1985

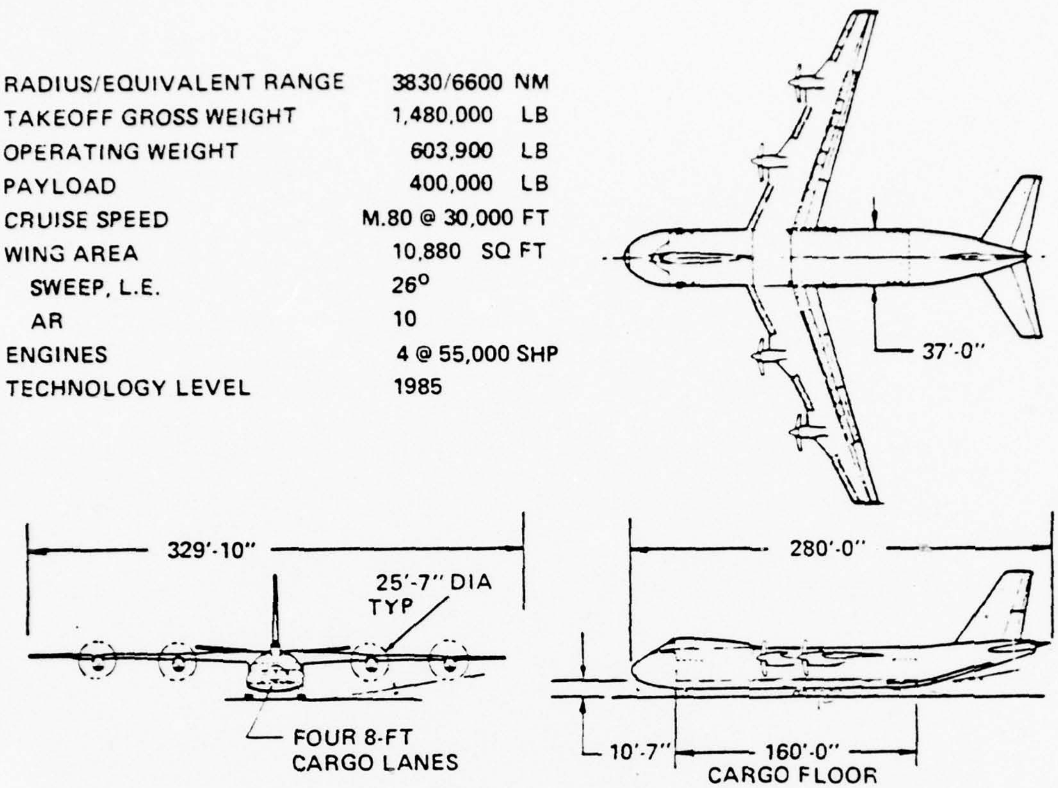


Figure 4-1 Model 1044-18 Propfan Airplane

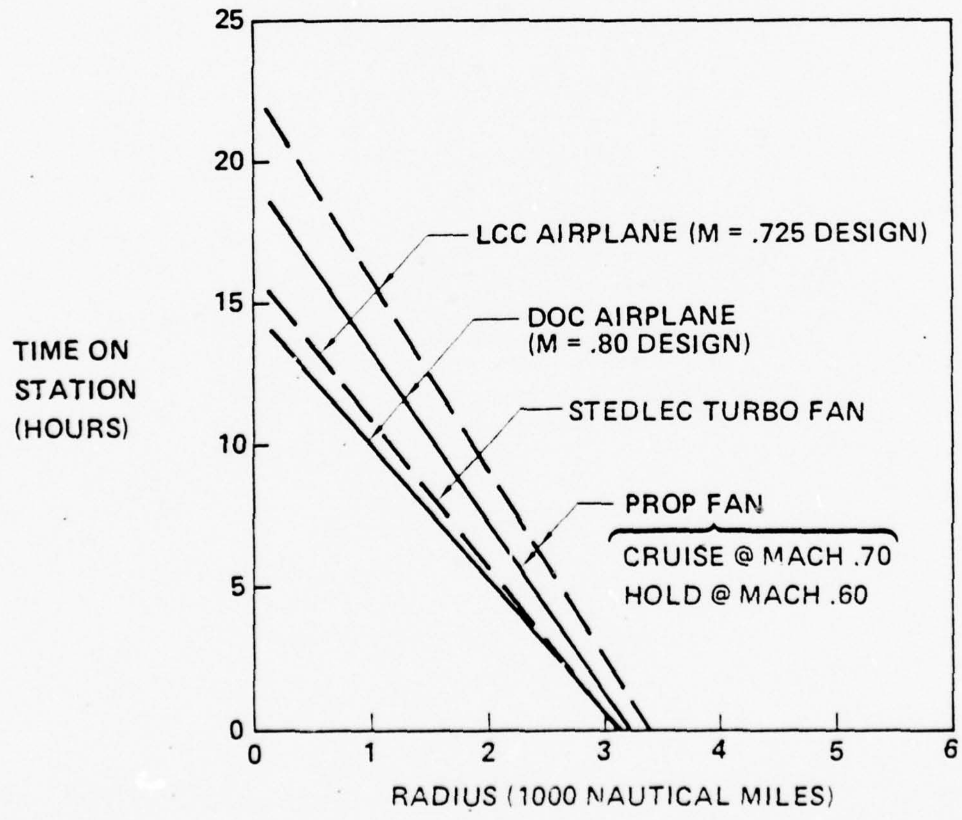


Figure 4-2 Advanced Technology Prop Fan Endurance

The 1985 technology implicit in the baseline included weight allowances for composite control surfaces. The Advanced Composite Design added to that: 1) wing and empennage primary structure, and 2) additional secondary structure such as nacelles. The configuration, showing those areas in which advanced composites are used, is shown in Figure 4-3.

RADIUS/EQUIV. RANGE	3,950/6,800	NM
TOGW	1,480,000	LB
OW	536,900	LB
PAYLOAD	400,000	LB
CRUISE SPEED M .81 @	35,000	FT
WING AREA	10,880	SQ FT
SWEEP, L.E.	26°	
AR	10	
ENGINES 4 @	81,800	SLST

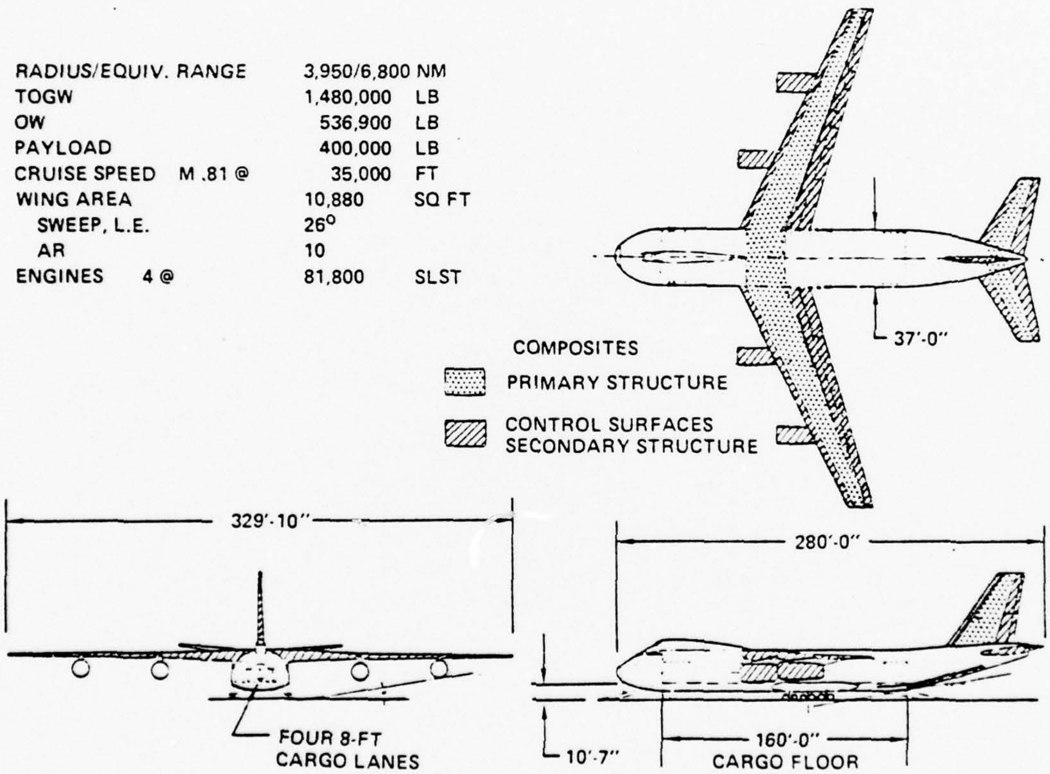


Figure 4-3. Model 1044-019 Composite Structures Airplane

The Boeing Company experience in advanced composite structures has indicated that weight savings can be realized as high as 30% over conventional aluminum primary structure. Figure 4-4 shows a Boeing designed and manufactured wing box section which had a savings of 30% in weight. Similarly, savings on the order of 15% have been realized in non-load carrying structures. In this analysis it was assumed that weight savings of 25% and 15% over conventional aluminum structure could be realized in primary and secondary structure respectively. The improvement in range amounts to about 13% over the 1985 baseline. The impact of advanced composites on gross weight is shown on Figure 4-5 for including the wing as well as the empennage as composite. The improvement is significant and amounts to a 6% reduction in gross weight.

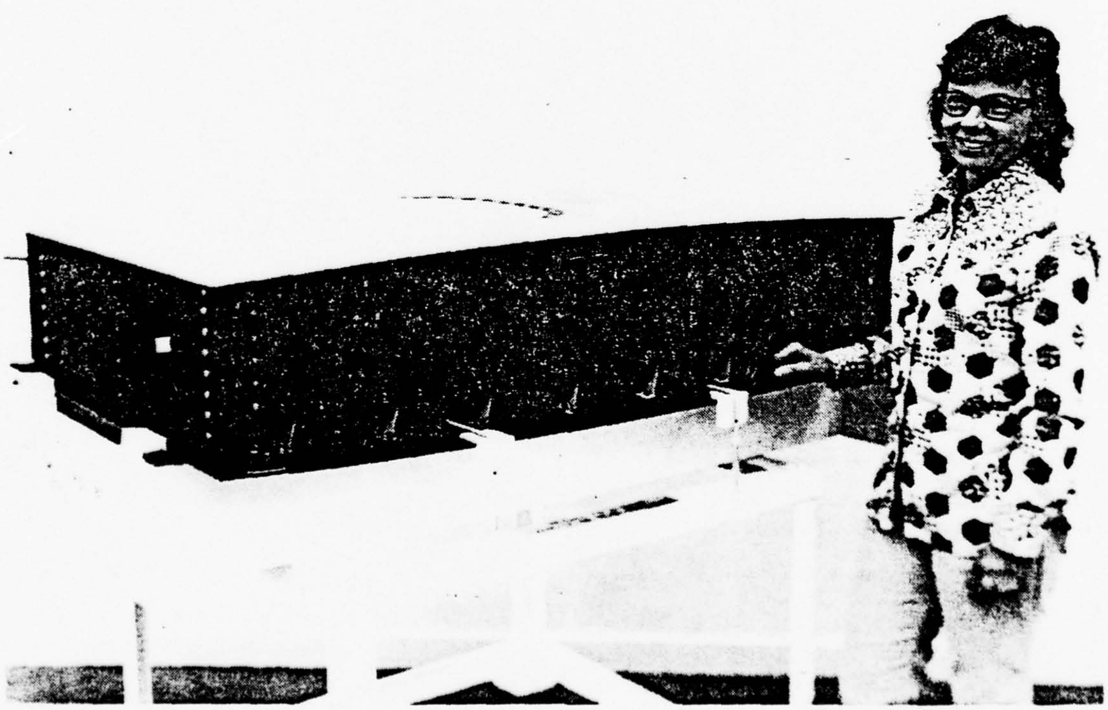


Figure 4-4. Wing Box Structure

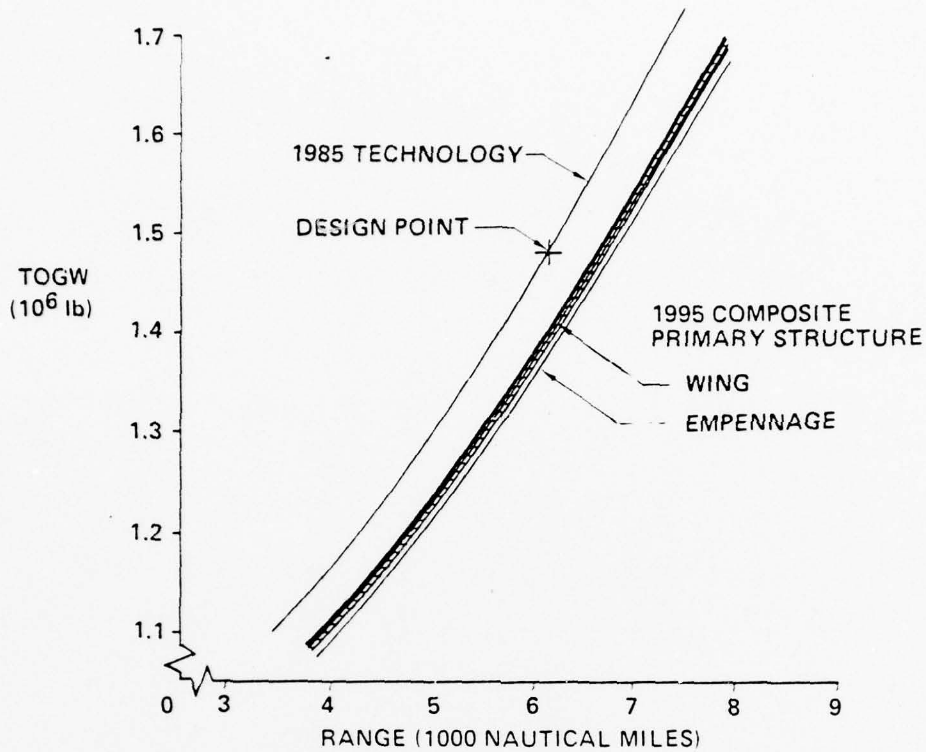


Figure 4-5. Effect of Structural Technology Take-Off Gross Weight

Figure 4-6 shows the impact of advanced composite technology on study DOC. Approximately 7% reduction in DOC is realizable at the design condition, assuming that no increase in airplane cost occurs as a result of the use of composite structure. In fact, most observers are in agreement that the costs *will be higher* although the labor level may be reduced. This will be discussed in Section 7.0.0 of this report.

The assumption that only the wing, empennage and secondary structure will be manufactured from advanced composites is conservative but represents a more realistic assumption than a 100% composite design.

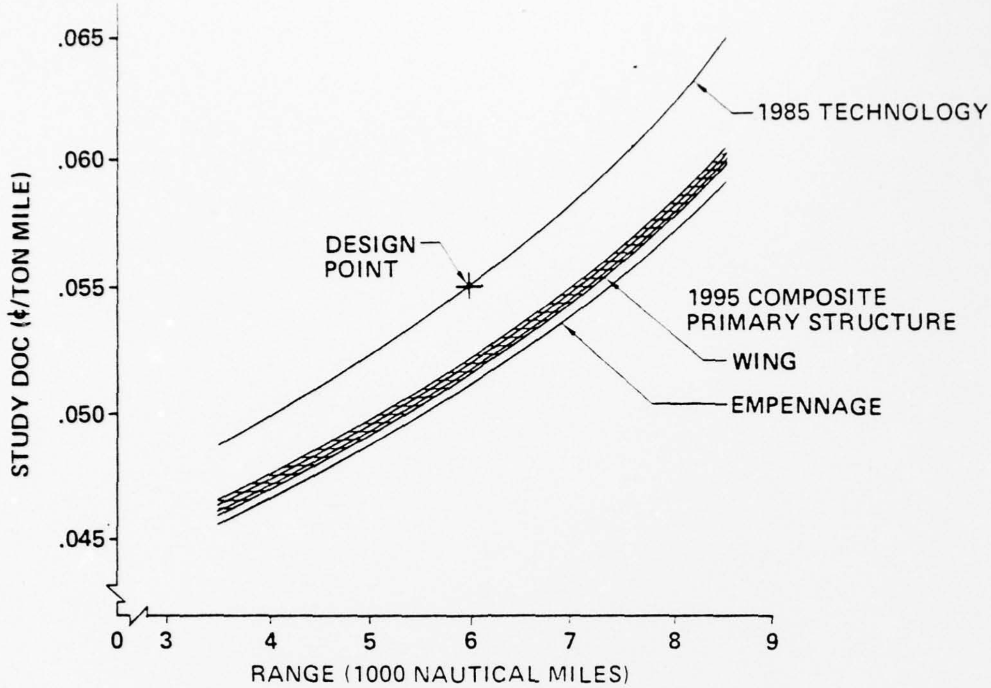


Figure 4-6. Effect of Structural Technology

4.1.3 Laminar Flow Control

Numerous studies and flight demonstrations, Reference 7, have indicated that a great potential for improved performance of long range aircraft can be achieved if the boundary layer can be converted to maintain a high degree of laminar flow with the resulting reduction in skin friction. Laminar Flow Control (LFC) is one of the areas undergoing intensive investigation by NASA under the ACEE program, and the Air Force Flight Dynamics Laboratory, Reference 8. The evaluation in this study is an adaptation of the preliminary results of Reference 8 to the mission and technology ground rules specified for this study. Specifically, in Reference 8 it was shown that LFC is shown to best advantage when introduced on a design utilizing a low wing loading because of the larger laminarized areas available.

The system envisioned, Figure 4-7, based on the work of Reference 8, incorporated four suction pump engines and four suction pumps distributed on the trailing edges of the wing, and one each in the aft body. Slots in the upper and lower surfaces of the wing and empennage surfaces provided the means by which low energy turbulent boundary layer air could be removed by suction, reducing the momentum flux in the boundary layer and providing for a laminar boundary layer over the wing.

At this stage in the work of Reference 8, the overall system weight of the suction system is undefined. However, a nominal value of 1.5 lbs/ft^2 of laminarized area was selected as being typical of the weight which might be expected. Laminar flow was assumed to exist over the forward 70% of the wing and empennage surfaces.

The LFC configuration is shown in Figure 4-8. It retains the essential features of the Baseline except for the change in the wing planform and increases the number of landing gear posts from 6 to 8 because of the increased gross weight. The wing plan form and cruise Mach number are those of the minimum fuel design discussed in Section 2.2.3 which approximates the plan form selected as optimum in Reference 8. An increase of 1250 nmi range, about 20%, results from the inclusion of LFC into the baseline design, based on the assumptions described above.

A comparison was made on the basis of radius of action and endurance between the turbulent flow minimum fuel design, and the design incorporating LFC. This comparison is shown on Figure 4-9.

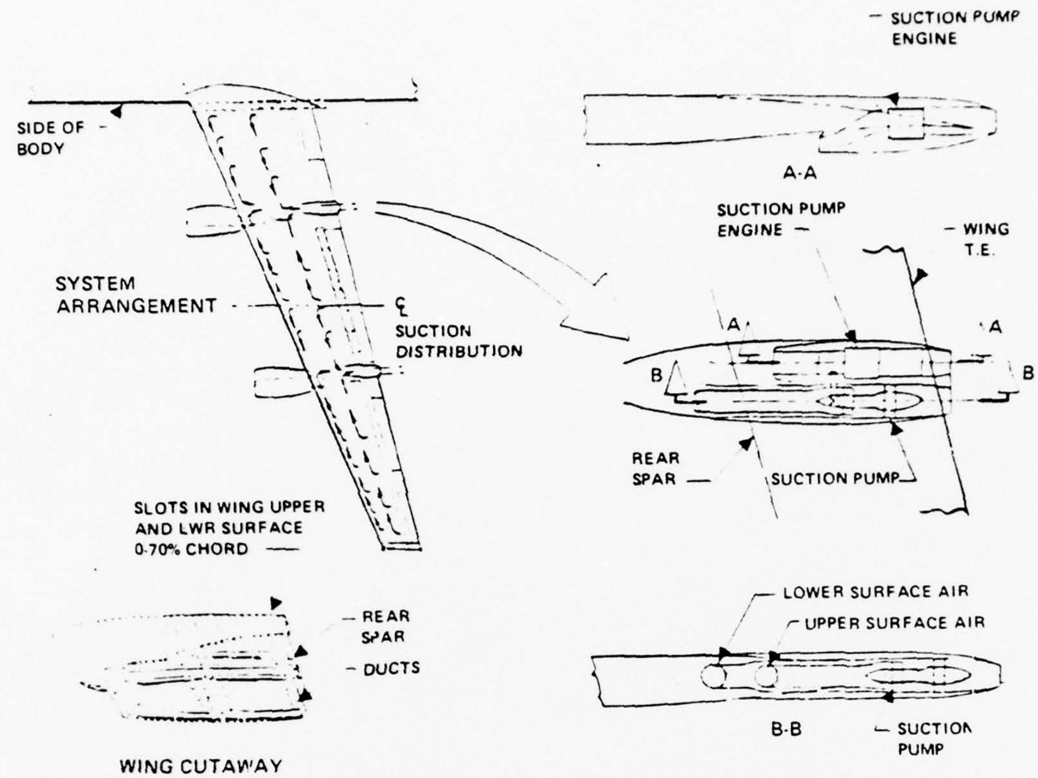
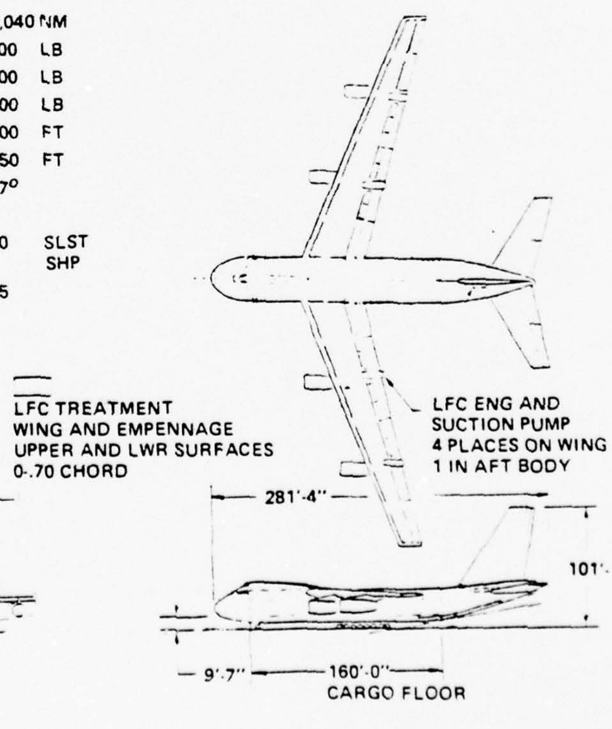


Figure 4-7. LFC System—Wing

RADIUS/EQUIV. RANGE			4,085/7,040	NM
TOGW			1,615,000	LB
OW			765,800	LB
PAYLOAD			400,000	LB
CRUISE SPEED		M = .78@	40,000	FT
WING AREA			16,150	FT
SWEEP, L.E.			22.7°	
AR			12	
ENGINES:	MAIN	4 @	74,300	SLST
	LFC	5@		SHP
TECHNOLOGY LEVEL			1985	



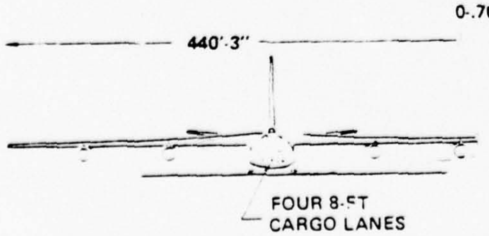


Figure 4-8. Model 1044-017 LFC Airplane

The benefits of LFC in terms of improved endurance and radius capability are substantial, about 20%, as shown in Figure 4-9.

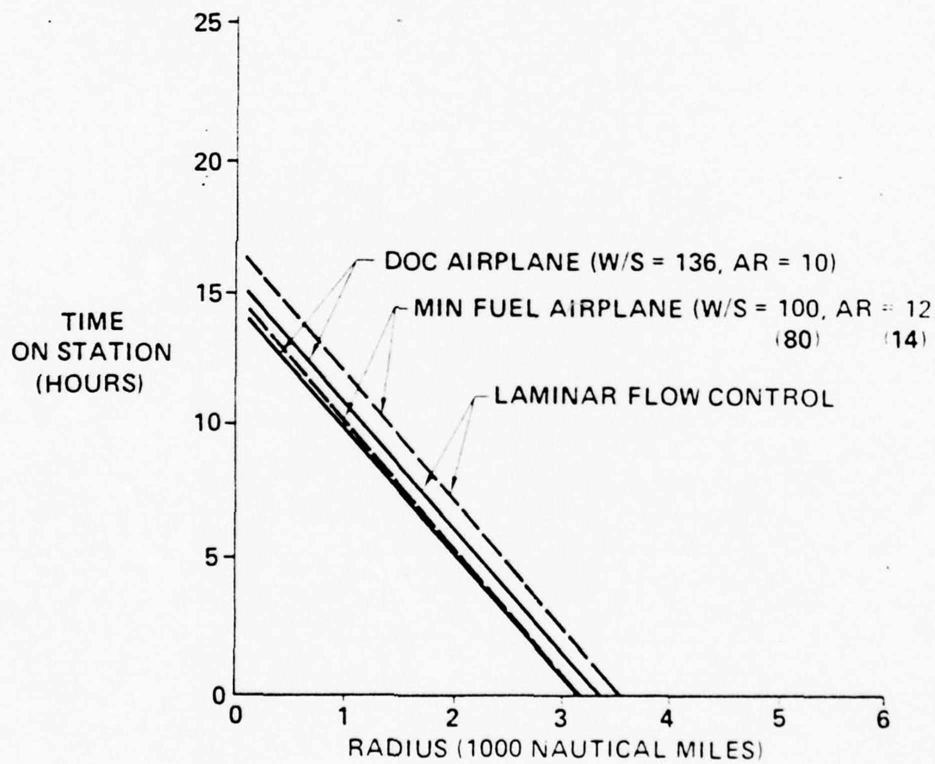


Figure 4-9. Impact of Laminin Flow Control on Endurance

4.1.4 Advanced Low Speed Aerodynamics

Considerable interest has existed in the use of low wing loading designs, which incorporate a simplified high lift system. These designs operate at lift coefficients at lift-off, considerably lower than those available through current high lift technology. This approach has several intrinsic merits, since a simplified flap system would provide significant improvements in weight and cost.

The take-off rules under which this study was conducted, MIL-C-5011A, did not provide for the additional requirements which are imposed by the ATA/FAR rules and therefore do not fairly evaluate the improvement available from low C_L, High L/D systems.

However, an attempt was made to evaluate the impact of operating at a lower C_L and the improved L/D associated with that C_L. The relationship between C_L and L/D for various flap technologies is that shown in Figure 3-49. The envelope of the curves was used for this analysis.

Figure 4-10 is a carpet plot showing the relationship existing between T/W and W/S and maximum takeoff gross weight for the baseline configuration. The baseline design point of 1.48 million lbs. is indicated which, using conventional flaps, achieved a C_L at lift-off of 2.0 and an L/D of 11.0. Reducing the C_L to 1.5 allows a lighter flap system and, for the same W/S, a lower gross weight, hence, shifting the carpet downward.

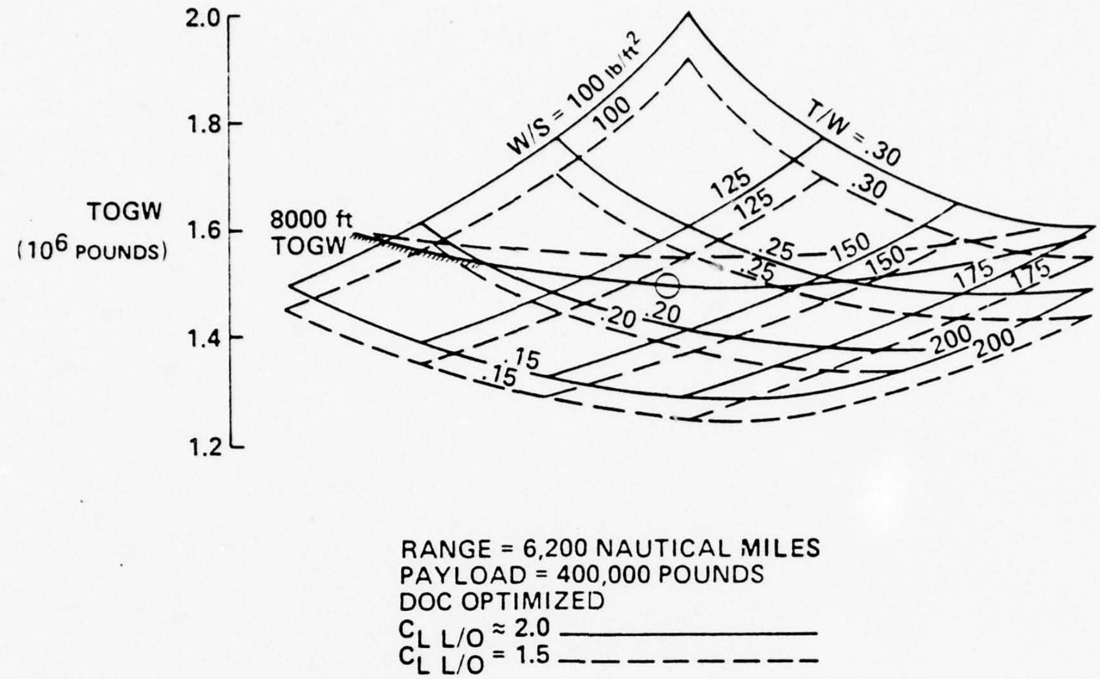


Figure 4-10. The Effect of Low Speed Aerodynamics

Superimposed upon the carpet in dashed lines is the T/W and W/S relationship resulting from using a simpler and lighter flap system, representative of a C_L level at lift off of 1.5 and an L/D of 15.0. Although lighter at given wing loadings, in order to achieve the 8000 ft take off, a lower wing loading, and/or a higher thrust/weight is required. The overall result is a takeoff gross weight approximately 3% heavier than if the more complex flap design were used. Hence the possible payoff of the approach necessarily requires a detailed examination of the effect of flap complexity on overall system acquisition and other costs, as balanced against heavier system weight.

Further, the real advantage of advanced low-speed, high-lift systems lies in reducing the severity of the constraint imposed by second segment climb requirements and is not reflected by Mil-C-5011A rules.

4.1.5 Boost Engines

As has been shown earlier, the baseline design is sized primarily by the takeoff constraint. The engine size, if matched for cruise, would result in a T/W significantly less than that which was required on the baseline. This reduction would impact the design and result in a significantly lighter gross weight. During the late 1960's an experimental boost engine, the XJ99, which had extremely high engine T/W - on the order of 15, was developed. This portion of the study is a preliminary evaluation of the use of such a boost engine to provide for a closer match between cruise and takeoff.

Four 15,000 lb boost engines were included in the baseline configuration mounted in the aft portion of the fuselage. The design was then resized. The carpet plot which illustrates the effect of the resizing is shown on Figure 4-11. As shown, the gross weight of the boosted configuration is slightly higher than that of the baseline. However, the takeoff constraint is substantially reduced and provides for a gross weight 100,000 lbs lighter than the baseline, while at approximately the same wing loading. A significant point is that the reduced T/W results in engine sizes which are in the size range which can be achieved by growth versions of the current generation of high by-pass ratio engines.

As in the case of the previous evaluation additional analysis of the cost impact due to increase propulsion system complexity, versus the reduction in aircraft weight, is needed to further evaluate this innovative concept.

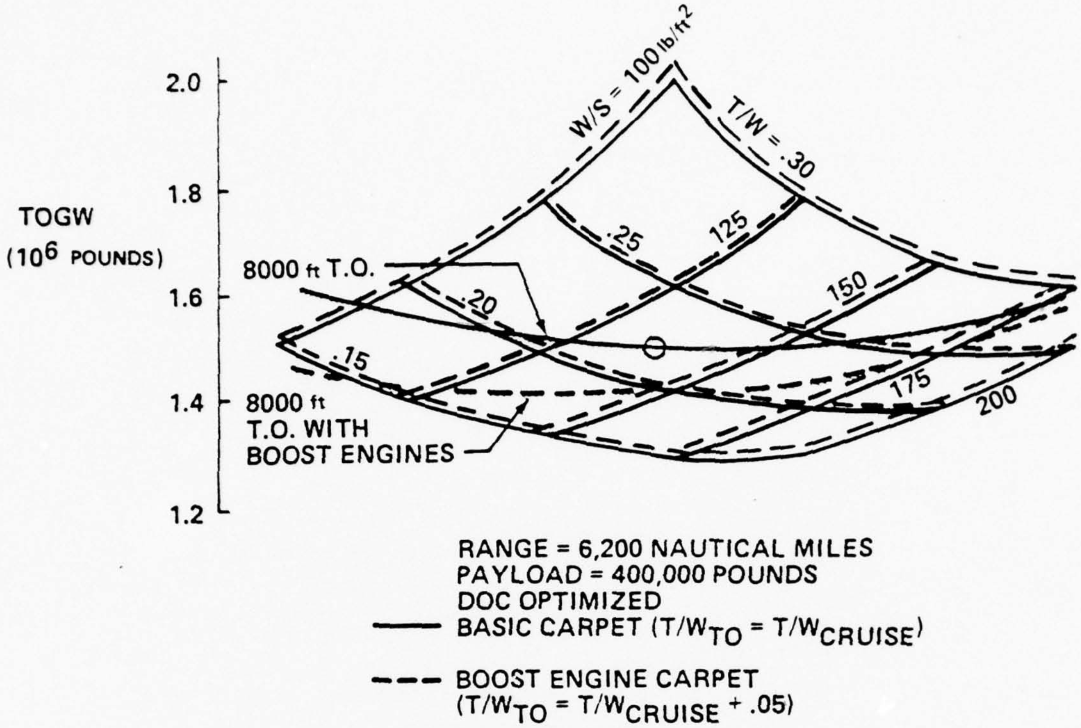


Figure 4-11 Advanced Design; The Effect of Boost Engines

4.1.6 Strut Braced Wings

Although identifying a strut braced wing as innovative may seem somewhat inverted, in fact the high aspect ratios and the resulting L/D which can be achieved with strut bracing of the wing of the baseline are attractive. In addition, it is an improvement which can be instigated in the near term.

Recent studies, Reference 9, have shown that strut configurations can be designed which avoid nacelle-strut interference and which have minimum interference drag. A typical strut configuration is shown in Figure 7-2, a configuration devised for cryogenic fuels, and is included here merely for illustrative purposes.

The impact of strut bracing the wing on endurance is shown on Figure 4-12 showing the radius of action to be increased by 200 nmi or 7%. Additional analysis of this concept is discussed in Section 7.0.

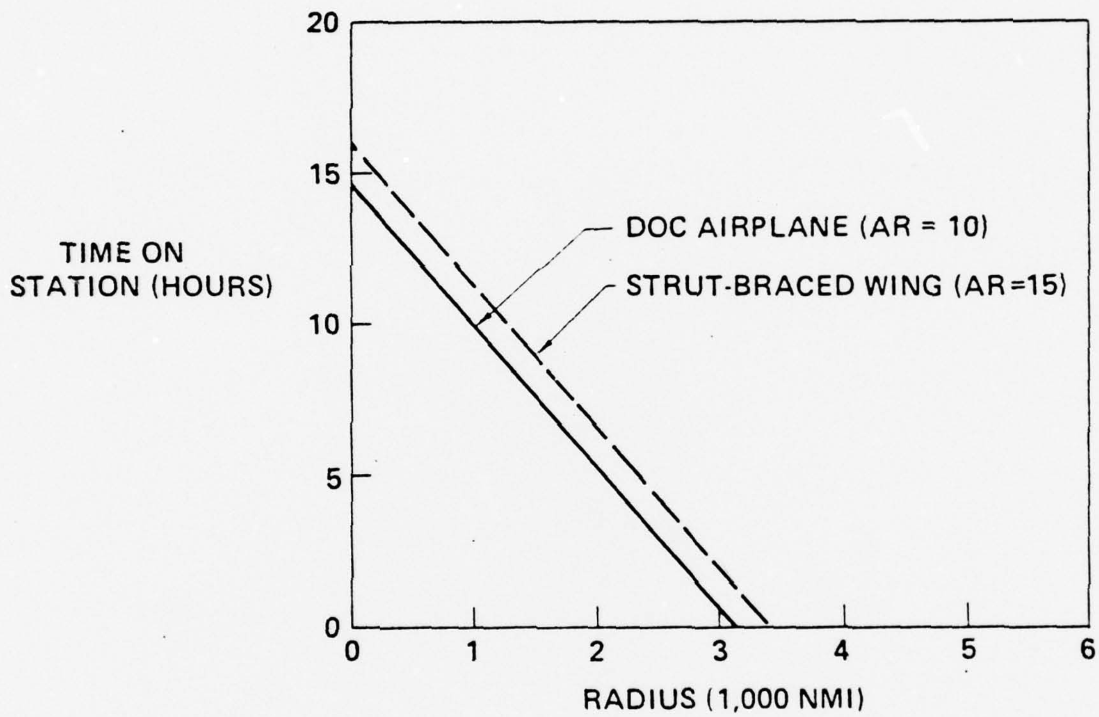


Figure 4-12 Advance Design Sensitivity: Strut Braced Wing

4.1.7 Air Cushion Landing System

Sufficient data was not available to provide other than a feasibility analysis of the use of an air cushion landing gear on the baseline configuration. One of the major questions remaining is the durability of the cushion skirts.

Figures 4-13 through 4-16 show design sketches illustrating an innovative design approach to utilizing ACLS on the baseline. In this approach the air bay skirt is deployed during landing by a cantilevered outrigger beam which rotates outward and locks in place for landing. Retractable supports are also deployed after landing to provide stability during loading.

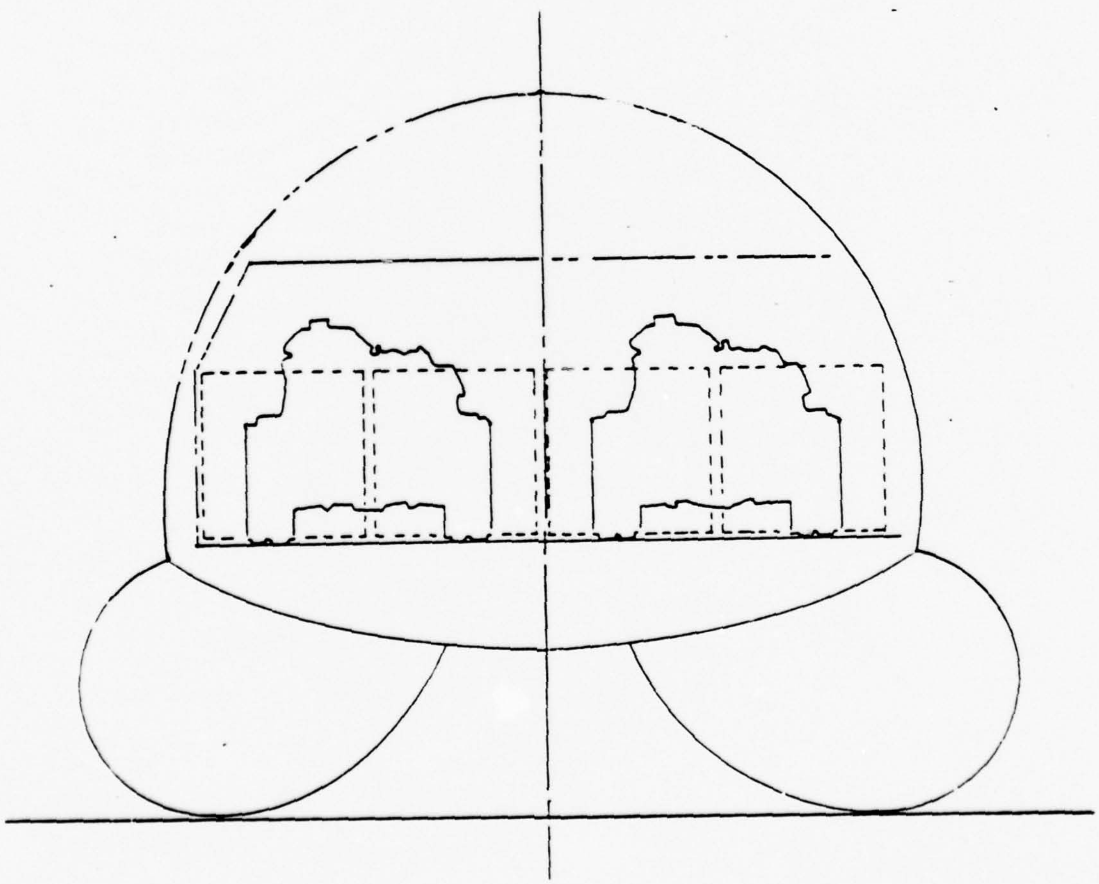


Figure 4-13 Air Cushion Landing Gear: Toroidul Trunk Configuration

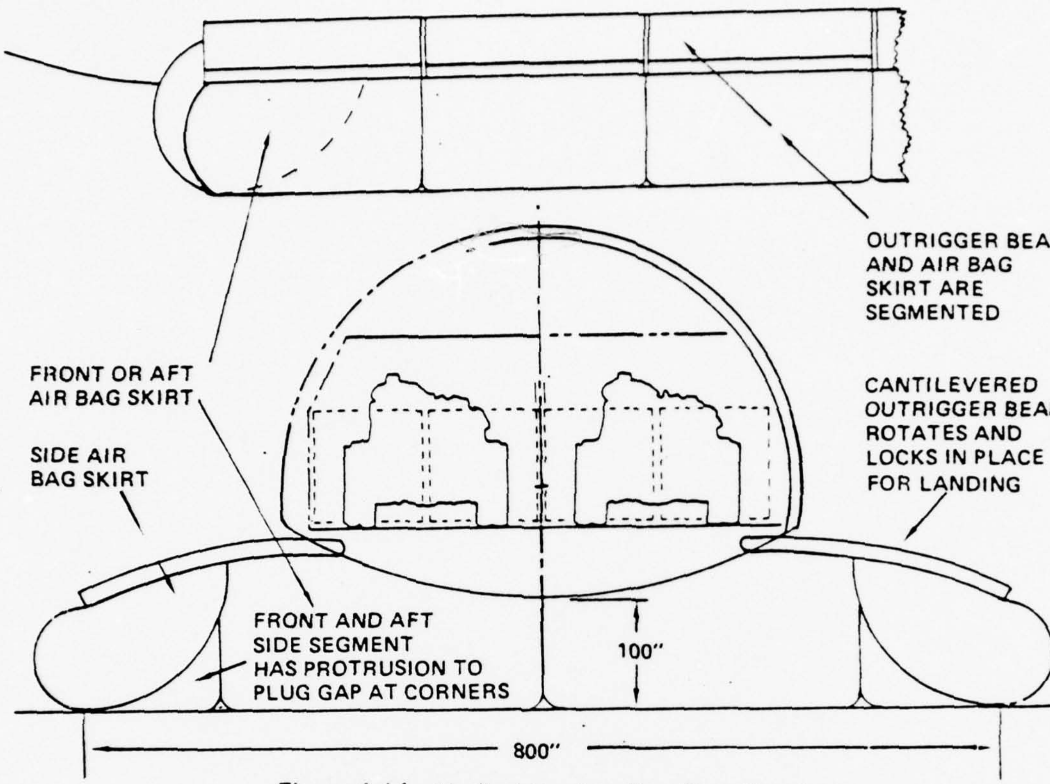


Figure 4-14. Air Cushion Landing Gear Outrigger

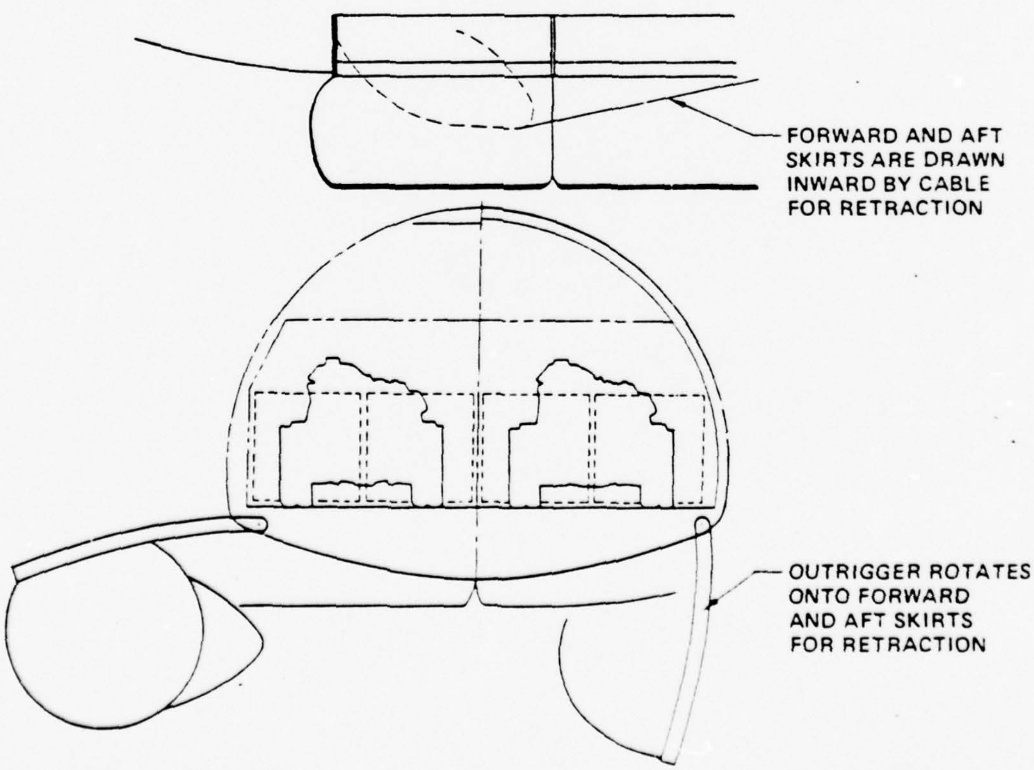


Figure 4-15. Air Cushion Landing Gear Method of Retraction

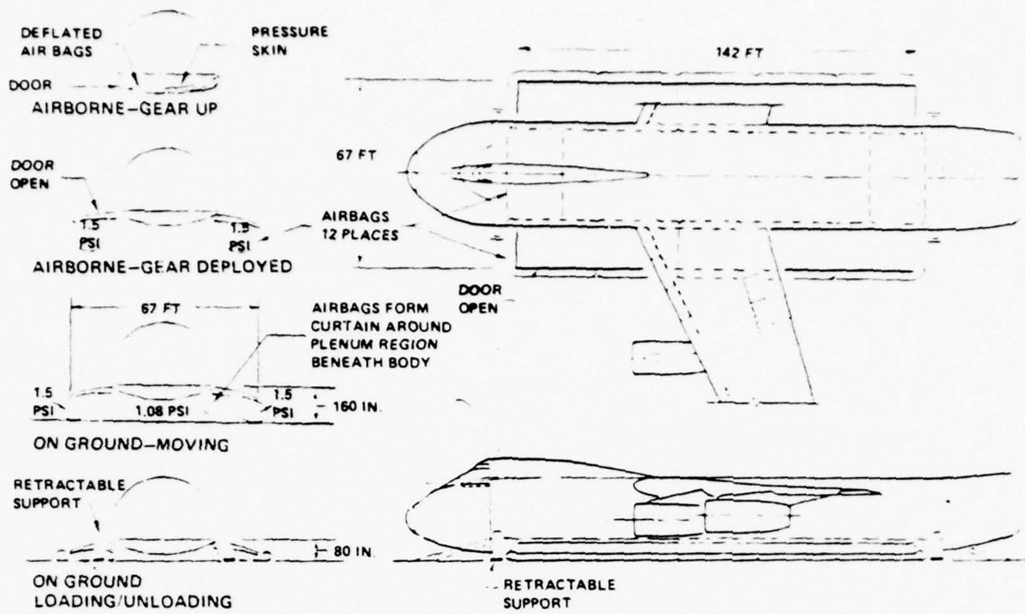


Figure 4-16. Air Cushion Landing Gear – Baseline Installation

4.2.0 MISSION SENSITIVITY

Mission sensitivity is defined for this study as the capability of the baseline design to perform missions requiring other payloads, ranges and field lengths, as distinguished from varying the design characteristics to match variations in mission.

4.2.1 Payload Range

The classic variation between payload and range is typical of mission sensitivity and is shown in Figure 4-17. The constant payload portion of the curve is determined by the 2.5 g maneuver load factor and the structural capability of the design. The second segment of the curve is the trade between fuel and payload at contant gross weight of 1.48 million lbs. At the junction of the second and third segments of the payload range curve, the trade of payload for fuel reached the point where maximum fuel capacity has been reached. Beyond this point, the off-loading of additional payload reduces the gross weight of the airplane (fuel at maximum capacity), thus accounting for the small increment of range improvement. The ferry range is shown as 12,500 nmi at zero payload.

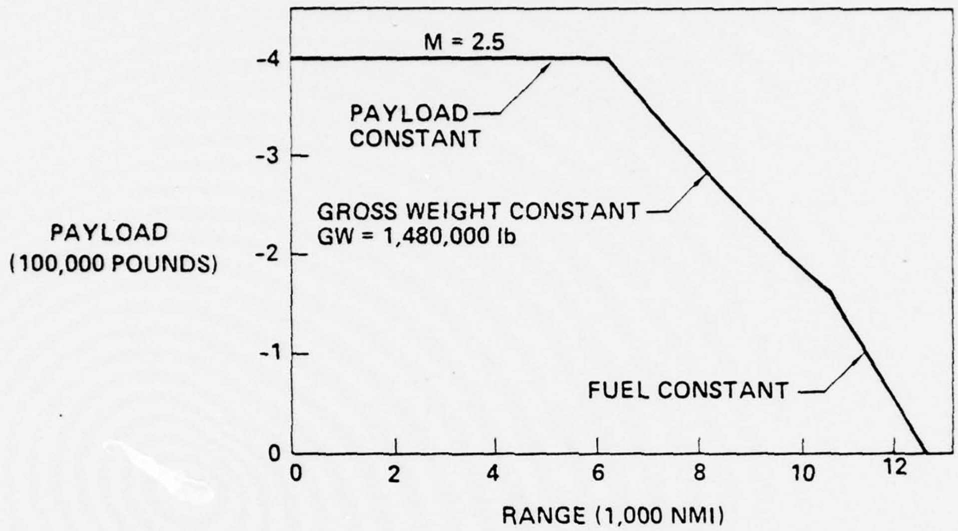


Figure 4-17. Mission Sensitivity — Payload/Range

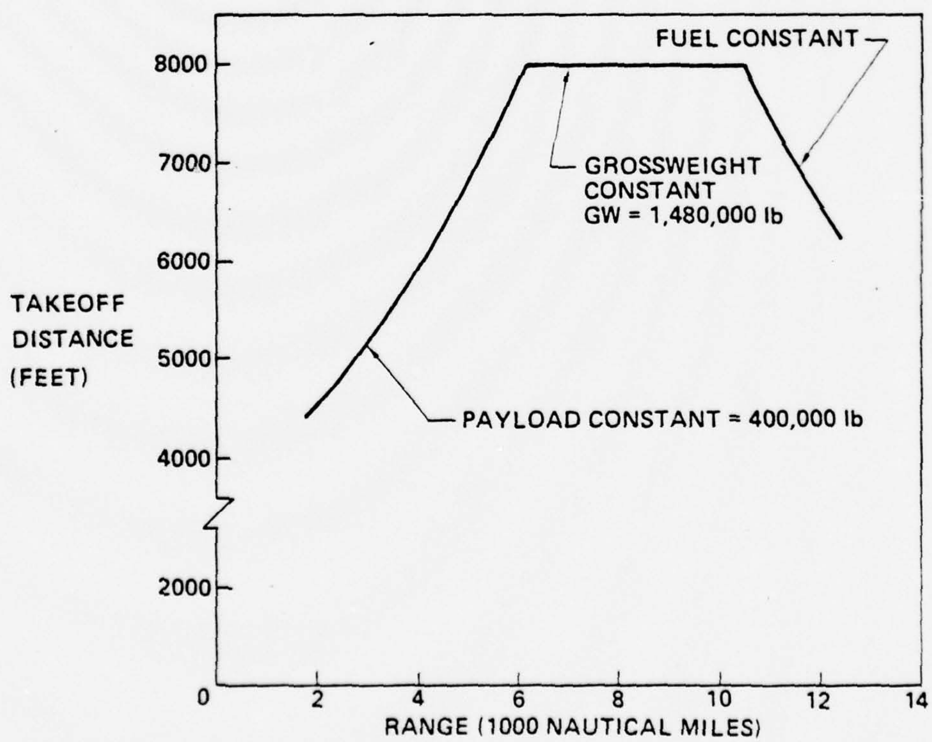


Figure 4-18. Mission Sensitivity — Takeoff Performance

4.2.2 Takeoff Field Length

Figure 4-19 shows the variation of Mil-C-5011A takeoff distance with the constant payload, constant gross weight and constant fuel portions of the payload range relationships shown on the previous figure. Of particular interest is the fact that at short ranges, on the order of 1000 nmi, takeoff field lengths as low as 4000 feet can be achieved with 400,000 lbs of payload. Appendix D shows the related fuel burn and block times.

An interesting comparison is made on Figure 4-20 where the mission capability of the C-5A is shown in terms of payload and fuel efficiency indicating an increase of 50% is available.

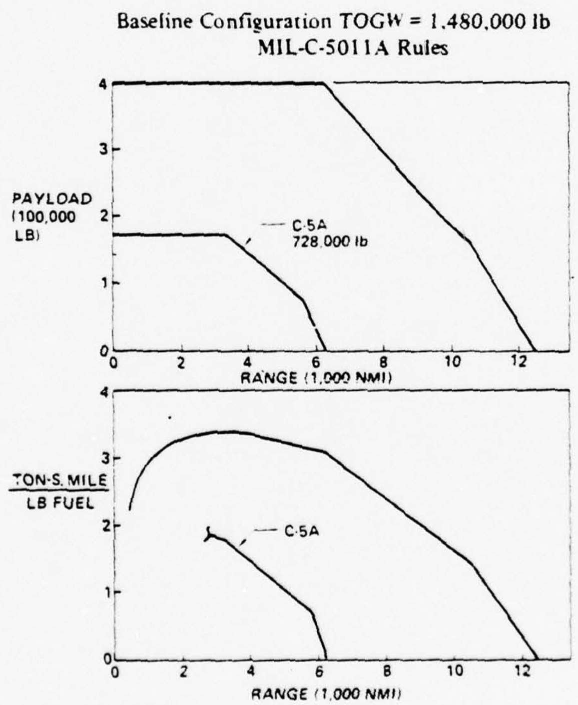


Figure 4-19. Comparison of Baseline with C-5A

5.0 Alternate Military Missions

- Strategic Ballistic Missile
  - Aft deployment
  - External Carriage
  - Vertical deployment
- Strategic Cruise Missile
- Tactical
- C^3

5.0.0 ALTERNATE MILITARY MISSIONS

A major interest in this study was the examination of the application of an aircraft, designed as an optimum military transport, to missions other than that for which it was primarily intended. The application to commercial freight transportation will be discussed in Section 6.0.0. This section will discuss the applicability to strategic offensive, tactical, and command, control and communication (C^3) missions.

5.1.0 STRATEGIC OFFENSIVE APPLICATIONS

5.1.1 Ballistic Missiles

USSR capability to achieve CEPs significantly reduced from first generation ICBMs has generated a concern for the survivability of US land-based ballistic missiles. As a result, alternate modes of basing have been considered, focused by Air Force studies on the MX missile system. MX studies have included options for land-based mobile and air-launched mobile basing. This section of the study had as its objective the examination of the feasibility of launching a ballistic missile, of the class considered for the MX, from a logistics transport, such as the baseline design, considering the different modes of launch and the penalties involved.

The mission scenario for strategic offensive launch is postulated to be: 1) fleet launch on warning; 2) loiter at radii less than 1000 nmi; 3) launch of missiles and return to base; or 4) return to base.

The missiles which were considered are based on Boeing MX studies and are shown in Figure 5-1. The baseline mission chosen for this study was the 180,000 lb, 90 inch missile.

Three launch methods were considered:

1) Aft egress with parachute extraction
2) Downward ejection through a missile bay
3) External carriage

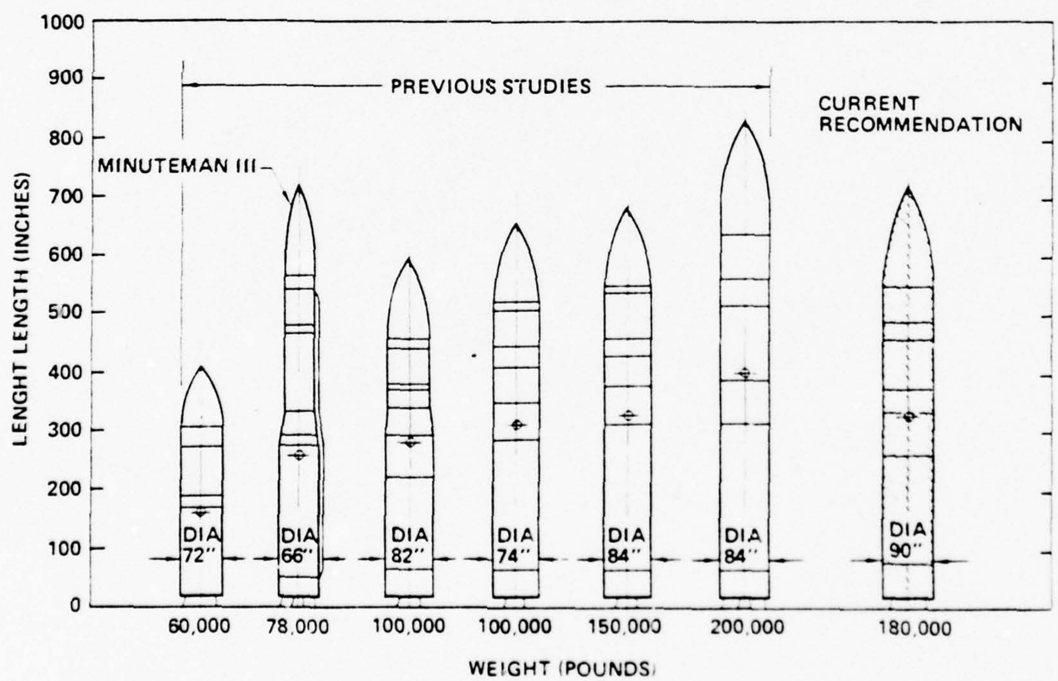


Figure 5-1 Candidate M-X Missiles – Air Mobile

5.1.1.1 Alternate Mission Performance

The performance of the alternate mission is measured primarily on the basis of endurance or time on stations. Figure 5-2 shows the variation of time on station versus radius of action as a function of gross payload. Also shown on Figure 5-2 is the impact of optimizing the configuration to maximize loiter capabilities. As in the propfan discussion of section 4.1.1, the increase in time on station occurs because loiter is maintained at maximum $\frac{L/D}{SFC}$ where maximum range occurs at maximum $\frac{M\ L/D}{SFC}$. The loiter occurs at M = 0.6, where as cruise out occurs at M = 0.8.

DESIGN POINT AIRPLANE DESCRIPTION

	DOC OPT	LOIT OPT
W/S	136 LB/FT^2	132 LB/FT^2
T/W	0.221	0.212
ΛLE	26°	10°
t/c	.10	0.15
AR	10.0	9.3

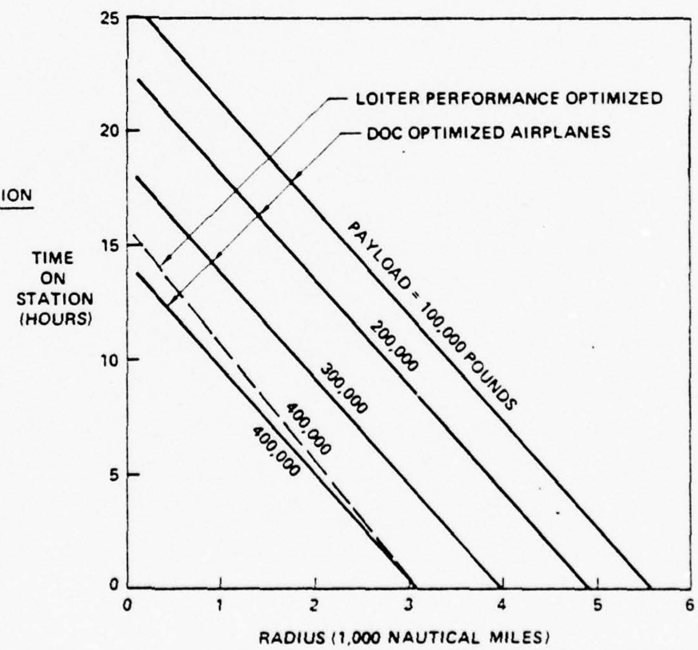


Figure 5-2 Loiter Performance — Alternate Mission Capability

5.1.1.2 Ballistic Launch Concepts

5.1.1.2.1 Concept I: Parachute Launch

As part of the MX demonstration program, a Minuteman missile was launched by parachute. After the missile was ejected and stabilized, first stage ignition was successful.

Figure 5-3 illustrates this concept as applied to the baseline configuration carrying two MX missiles. This concept adapts well to a highly common logistics configuration with an aft loading door.

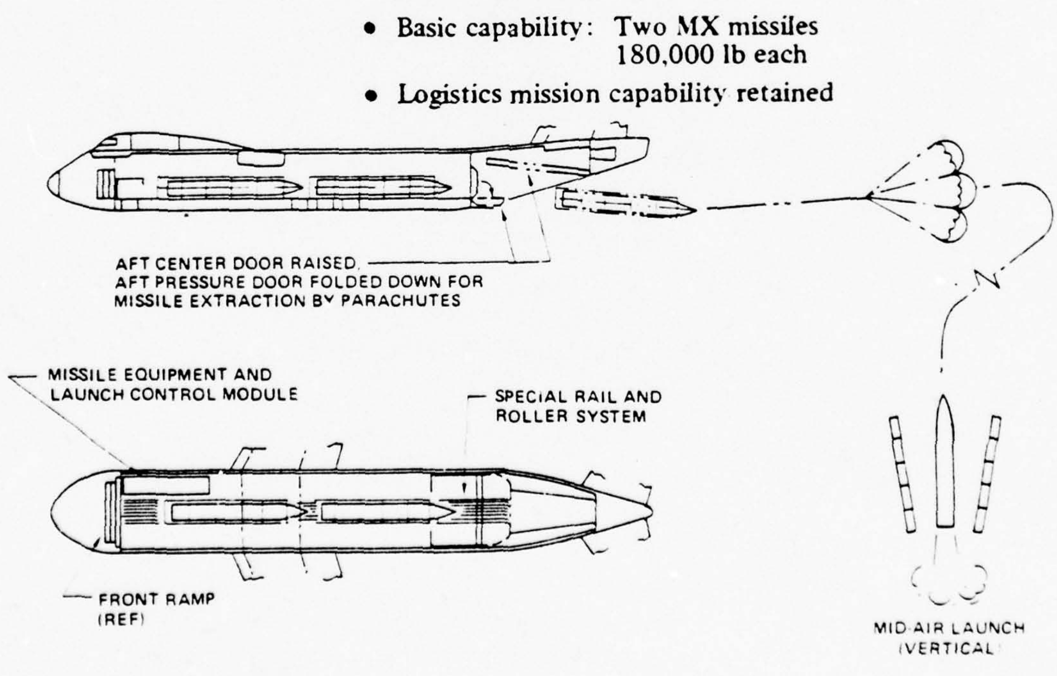


Figure 5-3 Ballistic Missile Carrier — Parachute Launch

5.1.1.2.2 Concept II: Wing Mounted

External carriage of two MX missiles appears to be a straightforward approach to missile launch. It is the most effective way of launching the missiles because of the initial ΔV which is imparted to the missile.

In order to make possible the carriage of missiles, hard points must be installed at the proper locations on the wing with the attendant increase in structural weight. An increment in drag must also be considered. Figure 5-4 illustrates this concept.

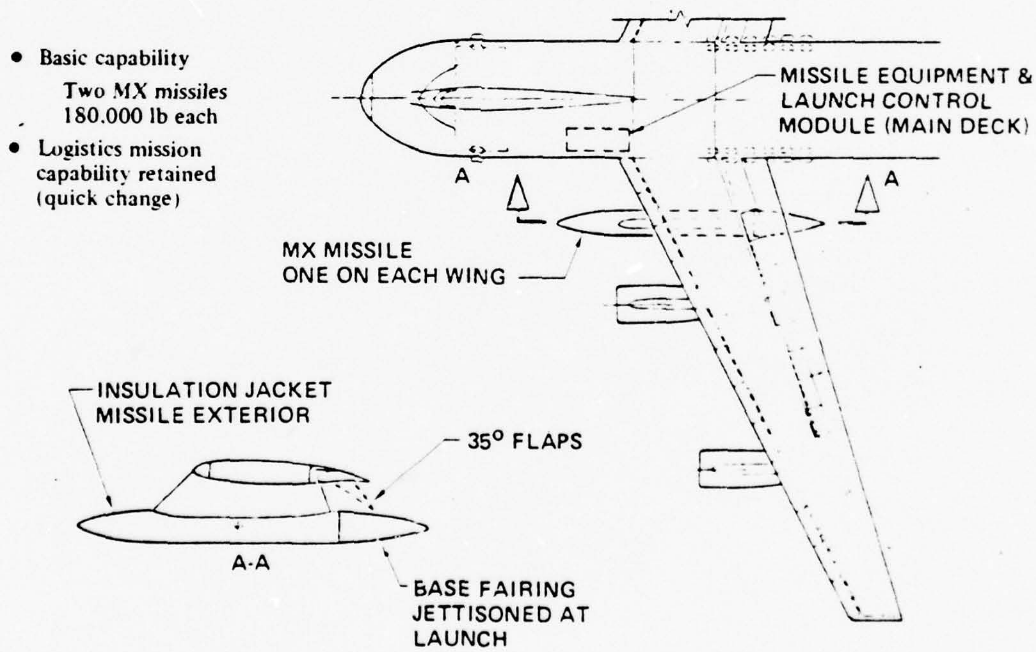


Figure 5-4. Ballistic Missile Carrier Wing Mounted

5.1.1.2.3 Concept III: Bomb Bay Drop

The concept which requires the maximum modification to the logistics transport design utilizes a bomb bay through which the missile is dropped. The missiles are guided to the bay by means of an overhead rail transfer system in combination with floor tracks. Figure 5-5 shows this approach conceptually.

- Basic capability: Two MX missiles
  180,000 lb each
- Derivative airplane. Aft cargo doors and ramp deleted.
  Bomb bay and doors added.

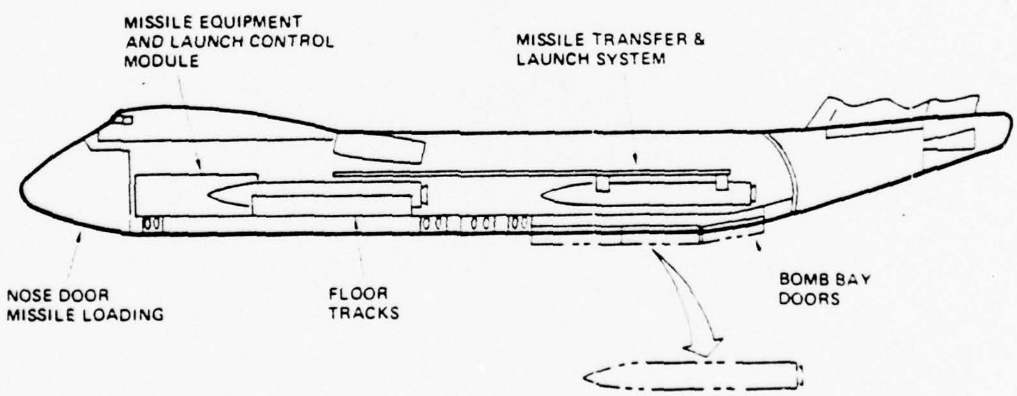


Figure 5-5. Ballistic Missile Carrier Bomb Bay or Drop

5.1.2 Strategic Cruise Missile

The concept of carrying long range cruise missiles in large subsonic aircraft is one which has been under consideration for a number of years as an approach to enhance bomber effectiveness by diluting the threat.

A deployment system was devised to allow the launching of Air Launched Cruise Missiles (ALCM) from the Baseline Aircraft. The concept involves installation of a small launch door on each side of the fuselage through which the missiles are ejected. Rotary launches similar to those used on the B-52/SRAM system are postulated. Figure 5-6 shows a schematic of the deployment system.

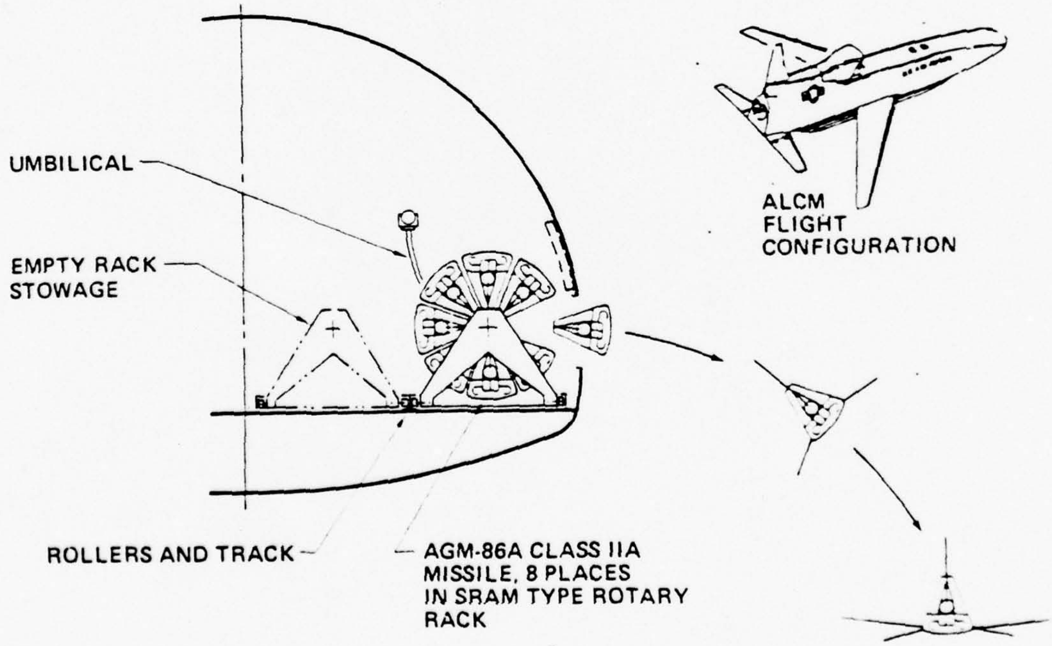


Figure 5-6. ALCM Launch Concept

The rotary launchers are mounted on tracks and are circulated aft to the launching station. After ejecting the complement of eight missiles, the rotary launcher is removed and stored forward. Another rotary rack is moved. to the launch platform and the launch sequence repeated. Figure 5-7 shows a configuration for the baseline design which will accommodate 18 rotary racks and 144 ALCMs.

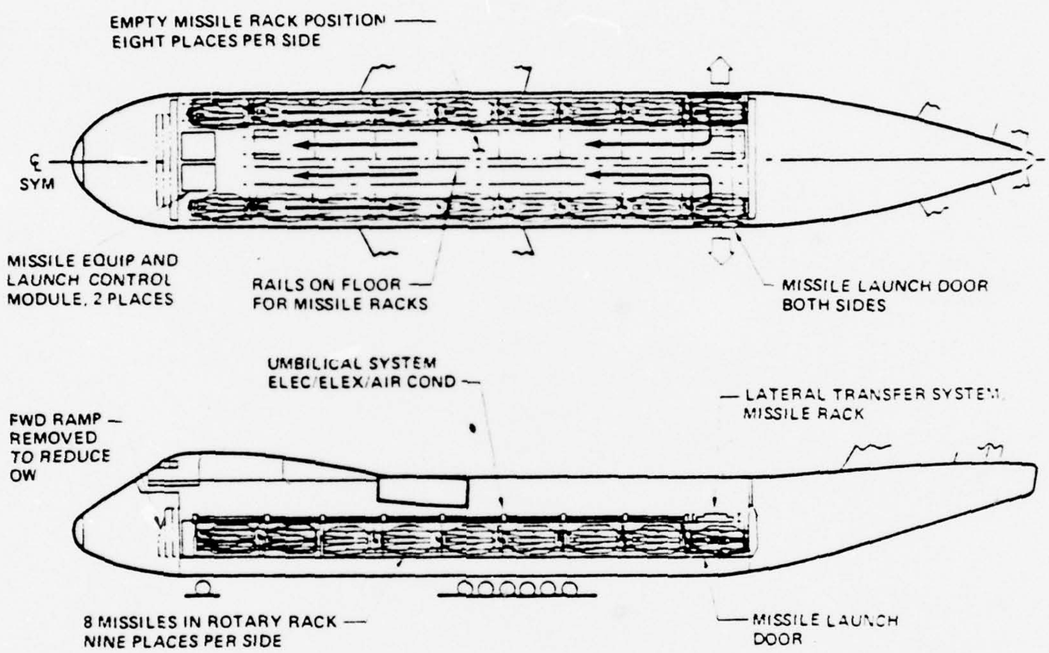


Figure 5-7. ALCM Carrier

The mission which is postulated for ALCM capability involves fly-out to a range of approximately 4,000 miles, loiter for sufficient time to launch all missiles and return to base; or not launch missiles and return to base. In order to accomplish the large radius of action, the payload can be reduced. Figure 5-13 shows the capability of the Baseline/ALCM system with 1/2 the possible complement of ALCMs. A reduced payload also reduces the attractiveness of the system as a high value target.

5.2.0 TACTICAL WEAPON SYSTEM

One of the particularly interesting aspects of the use of a large logistics transport in an alternate role is its use in transporting tactical weapons long distances rapidly. Such a system has been studied as part of the Microfighter concept, Reference 10. The concept involves the airborne deployment of a small, compact fighter type aircraft to provide instantaneous air superiority in areas where ground basing and support is not available.

Figure 5-8 shows a schematic representation of a full payload of 26 microfighters. Figure 5-9 shows a typical microfighter design. Two deployment bays are provided in order to enhance speed of deployment and to increase reliability of operation. The launch sequence relies on an overhead rail system to move the aircraft to the fore and aft launch stations. Each station has an airlock to allow the fighter bay to be convenient for maintenance personnel to carry out their function unencumbered by extraneous equipment.

In order to achieve a deployment to Europe and return, a reduced complement of microfighters would be required. Figure 5-13, shows the capability of the tactical derivative would be 12 microfighters deployed to Europe with the capability to perform various missions for seven hours and return to CONUS. Previous estimates have shown that approximately one sortie per airplane per hour is possible which would provide for 75 to 100 sorties per mission depending on the sortie characteristics.

A similar analysis was performed with the fighter capability being provided with F-16 fighters. Eight F-16 fighters could be deployed in a fashion similar to that envisioned for the microfighters and would provide a sortie capability proportionately less than that of the microfighters.

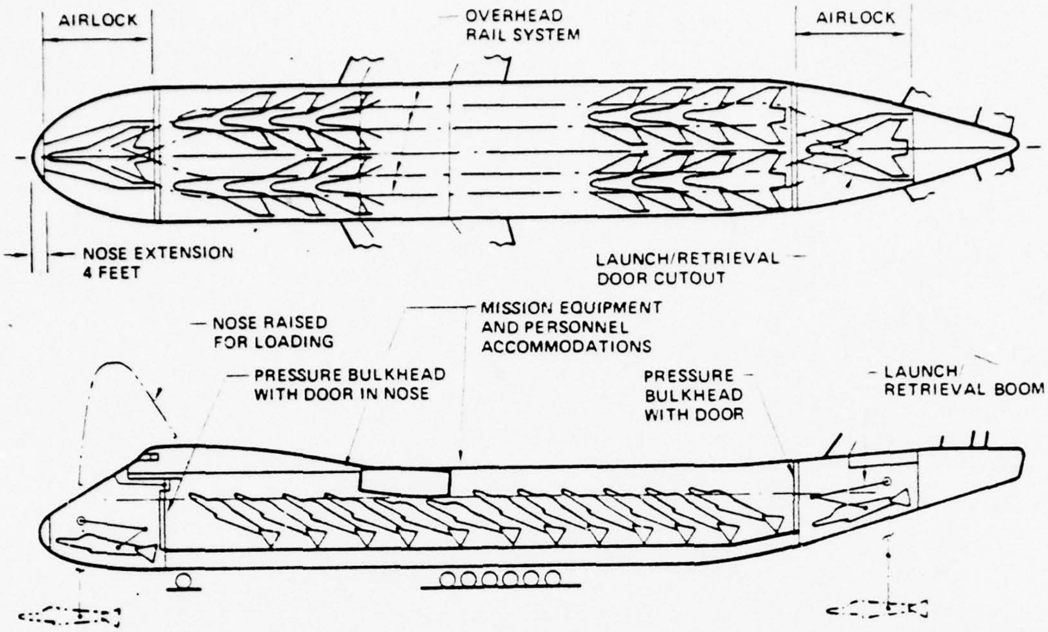
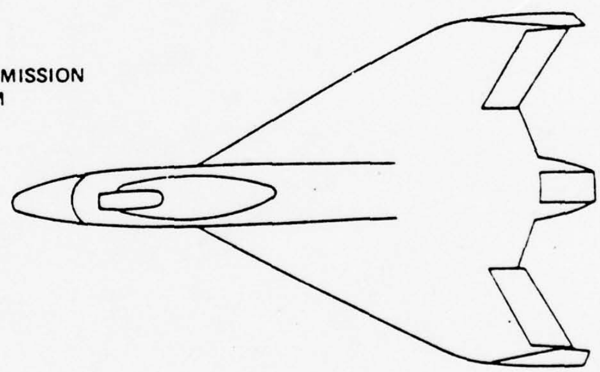


Figure 5-8. Tactical Carrier

GROSS WEIGHT	8,500 LB FIGHTER MISSION 12,500 LB MAXIMUM
OW	5,680 LB
PAYLOAD + FUEL	2,820 LB
WING AREA	200 SQ FT
ENGINE	11,200 SLST
ARMAMENT OPTIONS	25 MM GUN A/A MISSILES AIM-7 MISSILES SMART BOMBS



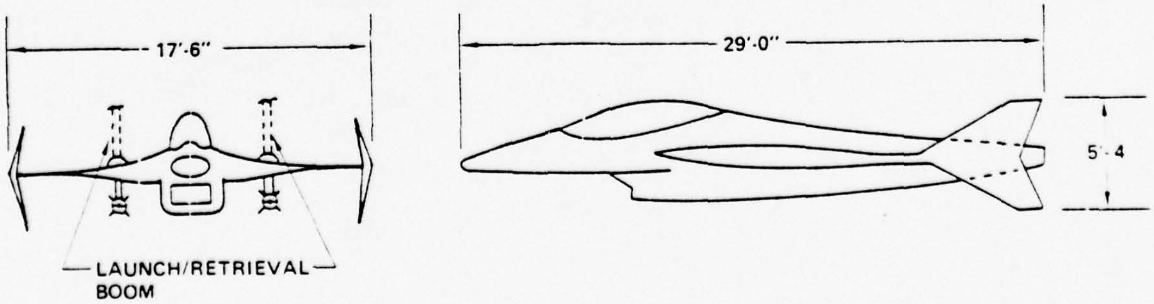


Figure 5-9. Microfighter

Figure 5-11 shows a conceptual configuration of a combined C^3 and Tactical/F-16 capability illustrating the mode of operation and installation of the Tactical/F-16 system. Certainly one of the advantages of the wide cargo bays selected for the baseline is the flexibility of missions and the broad range of payloads which can be carried, as amply illustrated by the Tactical/F-16 configuration.

5.3.0 COMMAND, CONTROL AND COMMUNICATION (C^3)

Utilization of derivatives of existing aircraft for C^3 missions has a precedence which makes consideration a derivative of the IADS Baseline a reasonable approach to fulfillment of C^3 requirements. More particularly, the E-3A and Advanced Airborne Command Posts were derivatives of the Boeing 707 and 747 respectively, as were the original Airborne Command Posts derivatives of the C-135.

The approach taken to evaluate the capability of the IADS Baseline to fulfill the C^3 function was to assess the volume and payload requirements of the E-3A and E-4B and determine how they might be combined into an integrated C^3 capability combining the functions of an Airborne Command Post and the E-3A, Figure 5-10.

Reference 11 had shown that a cheek mounted phased array surveillance radar was a viable advanced technology capability to be considered for the C^3 airplane.

A self defense capability was included as a possible alternative because the C^3 requirement provided sufficient additional payload and volume. The concept included two deployable F-16 fighters which could be cycled to provide self defense or tactical capability. A schematic of this concept is shown in Figure 5-11, and a three view of the F-16 in Figure 5-12 for reference.

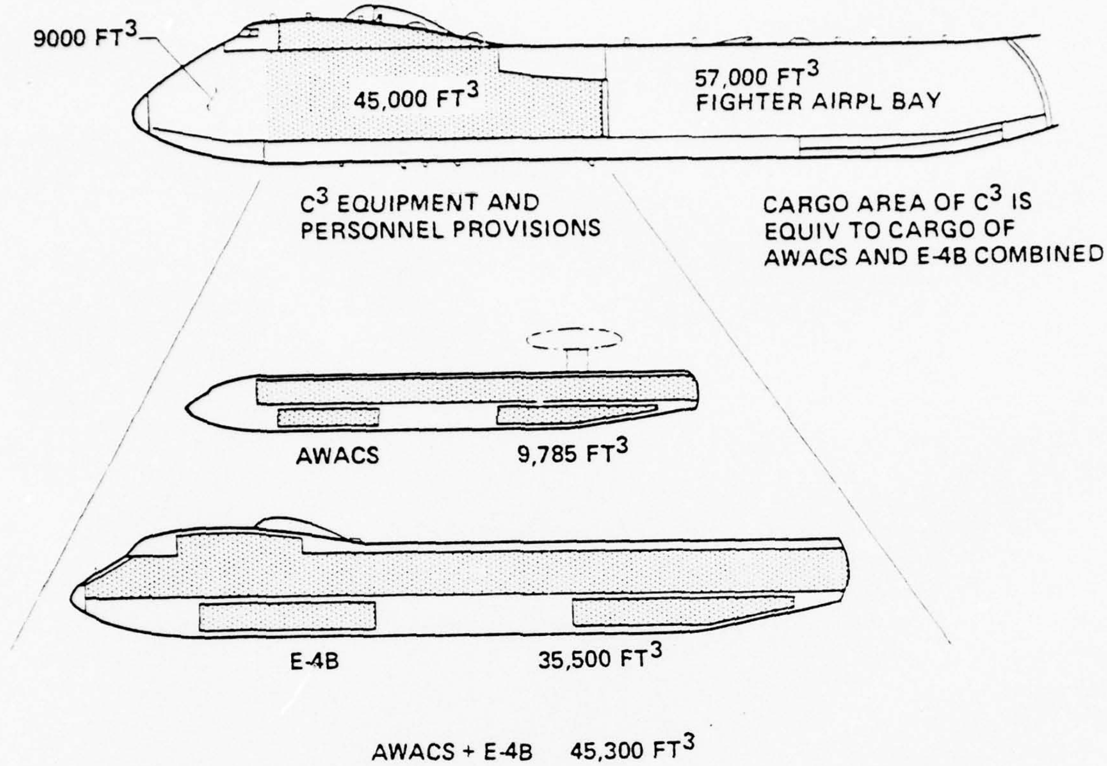


Figure 5-10. C^3 Volume Comparison

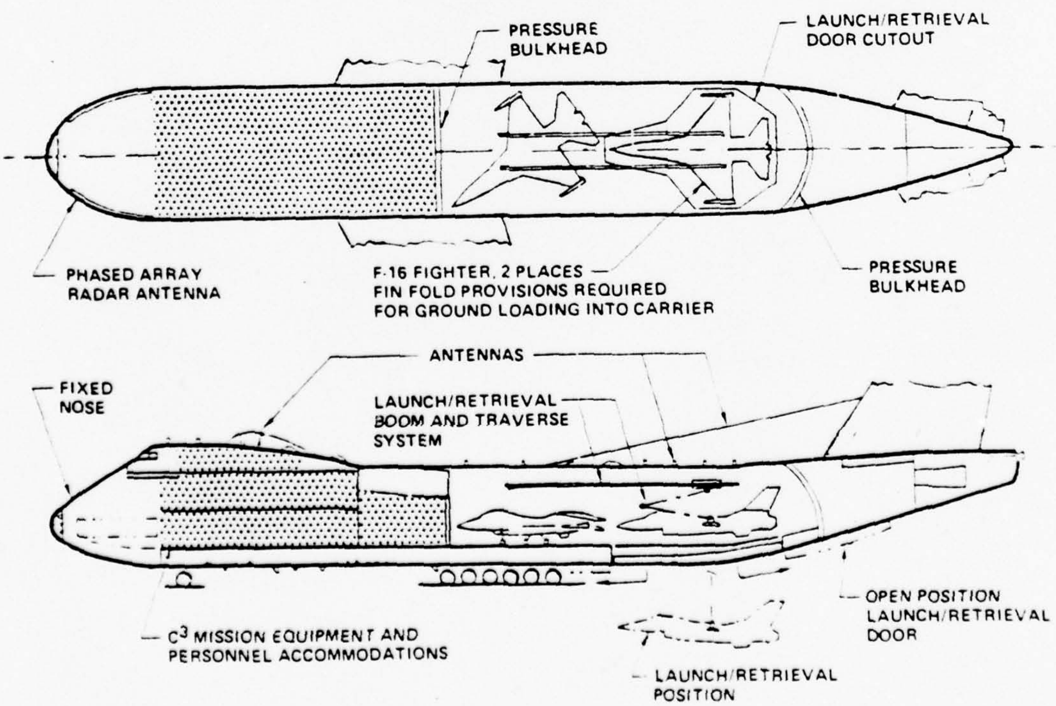
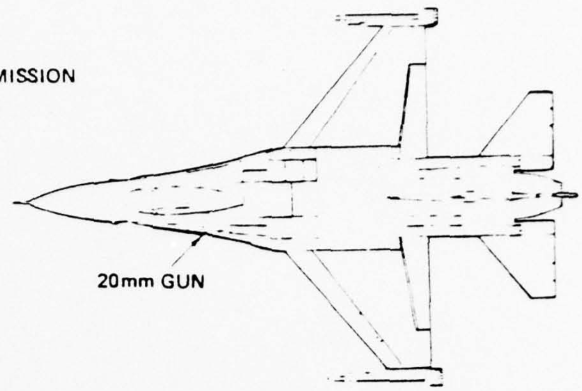


Figure 5-11. C^3 Mission Airplane

TOGW	22,200 LB FIGHTER MISSION
	33,000 LB MAXIMUM
OW	14,100 LB
INTERNAL FUEL	6,900 LB
WING AREA	300 SQ FT
WEAPON OPTIONS	20MM GUN A/A MISSILES A/G MISSILES BOMBS ECM POD SPECIAL WEAPONS



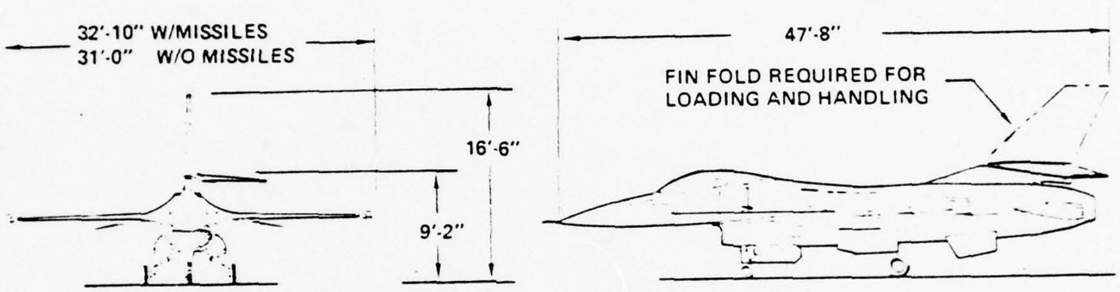


Figure 5-12. F-16 Fighter

The performance of the C^3 derivative is also shown on Figure 5-13. The operational concept would provide for deployment from CONUS plus seven to ten hours on station and return to base. The deployment of the command post alone would provide a capability of a day aloft orbiting over CONUS.

If the C^3 were configured for an ASW or sea surveillance and control capability times on station of 12 to 17 hours on station would be achievable at radii of 2 to 3 thousand miles.

DESIGN POINT AIRPLANE DESCRIPTION

	DOC OPT	LOIT OPT
W/S	136 LB/FT^2	132 LB/FT^2
T/W	0.221	0.212
Λ_{LE}	26°	10°
t/c	.10	0.15
Æ	10.0	9.3

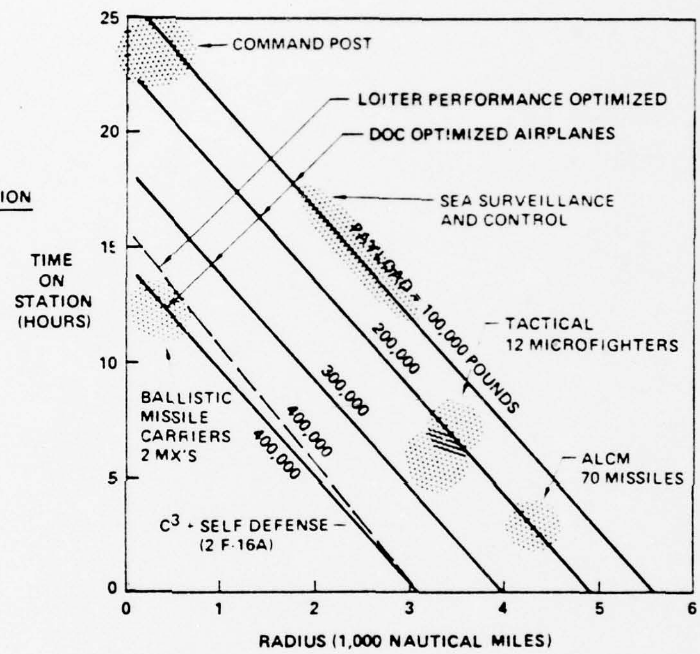


Figure 5-13. Loiter Performance

6.0 Commercial Commonality

1. Commercial baseline
   - Payload/range
   - Direct operating costs
2. IADS baseline
   - Payload/range
   - LCC
3. Commercial derivative
   - Derivative impact
   - Weight increments

6.0.0 COMMERCIAL COMMONALITY

The application of military transports to commercial use has been a goal for many years and has been attempted with varying degrees of success on a number of airplanes during the past several decades. Generally speaking, the design requirements imposed by military operations are such that a degradation of capability results relative to a competitive airframe, designed to commercial rules.

6.1.0 BACKGROUND

The recent interest in a commercial dedicated air freighter, an airplane designed specifically to carry air freight, has renewed interest in a transport design which would have a high degree of commonality between the military and commercial versions.

In 1974 MAC generated a concept development paper entitled the "Military Concept of the C-XX", Reference 12, which discussed the design requirements which MAC would like to see included in a new commercial freighter. This concept was premised on the assumption that a modern, efficient, commercial freighter would generate a sufficiently large market that a large number of freighters would be produced. Those freighters then would provide MAC with a major increase in surge capability through mutually beneficial arrangements between the commercial and military sectors as has previously been the case in the Civil Reserve Air Fleet, (CRAF).

Recent projections of market developments, Reference 13, have indicated that numbers of commercial airfreighters sufficient to provide the incentive for a major new commercial freight development program may not occur in the near future. In addition, the profitability of the Cargo Air-Carriers, an essential ingredient to a new development program, has not recently been strong because of increases in the cost of operations.

In order to generate the necessary demand, a high return on investment must be generated which will in turn allow a lower yield, or charge to the shipper, and thus encourage additional movement of airfreight. If a military development of an advanced transport could offer significantly lower direct operating costs, the application of that aircraft to the commercial market in the form of a commercial derivative may play on the elasticity of the market in such a way as to create a significant demand.

The purpose of this section of the study was to evaluate the compatibility of the military transport design, as characterized by a commercial derivative of the IADS baseline, against commercial needs; also, to evaluate the impact of advanced technology on direct operating costs and thereby assess the attractiveness of the commercial derivative by comparison to other commercial freighter designs.

The inverse problem, that of evaluating the commonality of a commercial freighter to a military derivative has been evaluated in Reference 13 and will not be treated in this study.

Commercial commonality involves many more issues more than determining the necessary weight increments which can be traded between the military and commercial configurations. Figure 6-1 schematically identifies some of the issues which might affect commercial commonality and all of which affect the design requirement; such as service life, flotation, need for drive through capability; the operational concept, the maintenance plan, cargo handling, and airfield compatibility.

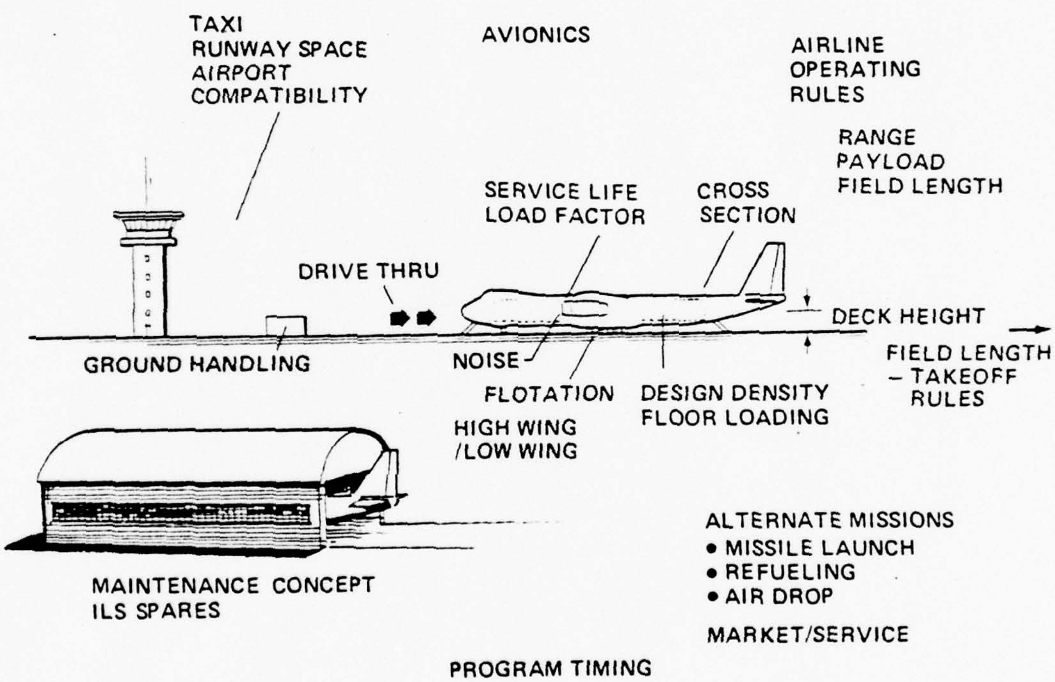


Figure 6-1. Commercial Commonality

6.2.0 IADS BASELINE

The baseline design was exercised on an ATA mission to determine the effect of commercial rules on the mission range. Generally, the commercial mission requires more reserve fuel than does the MIL-C-5011A rule.

6.2.1 Commercial Derivative

A commercial freighter was derived from the IADS baseline by removing all that equipment which might be easily removed. The philosophy for generating a commercial derivative was as follows.

1. The initial development was based on the need for a military capability and the development sponsored by and paid for by the military.
2. Commercial interest existed for a commercial freighter which might be based on a derivative of a military transport.
3. The commercial derivative was derived by removal of that military equipment and structure from the military transport which could be easily accomplished and did not entail major structural modification. That equipment could then be easily and quickly reinstated in time of emergency. Both the military transport and the commercial derivative might well be built on the same production line with kit changes from one configuration to another.

Figure 6-2 shows those items which were considered as being easily removable. While not being exhaustive, the list is representative of the magnitude of weight which might be gained by conversion and is in general agreement with the studies of Reference 13 which considered a military derivative of a commercial freighter for approximately the same payload level.

Remove military cargo floor	−17,100
Remove tiedowns and rails	− 4,260
Remove forward ramp	−27,900
Remove aft ramp	−27,570
Remove military cargo handling equip	− 1,750
Add commercial cargo handling system	+ 16,000

Total operating weight increment = −62,580 lb

Ground Rules:

- Only easily removable structure and equipment are considered.
- Military specifications used for design of structure, systems, and equipment are unchanged.

Figure 6-2. Commercial Derivative Weight Increments

The major changes in the configuration occur because the fore and aft ramps are not needed for commercial operations, and the floor which for the military transport is designed for greater than 20,000 psi axle loads, can be replaced by a lighter weight commercial floor designed for containers. Additional study might well reveal additional items which might be removed. Also, relaxing the "easily removable" ground rule would result in additional weight saving. As an example, Reference 13 was able to reduce the "scar weight", or the incremental weight which a commercial freighter would carry in order to facilitate installation of equipment providing a capability to carry military vehicle, to the order of 5000 lbs. A typical example of such an optimized commercial freighter is shown in Figure 6-3, Reference 14.

MODEL 754-172P

ENGINES	CF6-50 (4)
THRUST	51,000 lbs
T.O.G.W.	1,079,400 lbs
O.E.W.	425,000 lbs
MAXIMUM PAYLOAD	415,300 lbs
WING AREA/AR	9,250 ft/10.5
CRUISE SPEED	M = 0.74
CARGO COMPARTMENT Δp	7.5

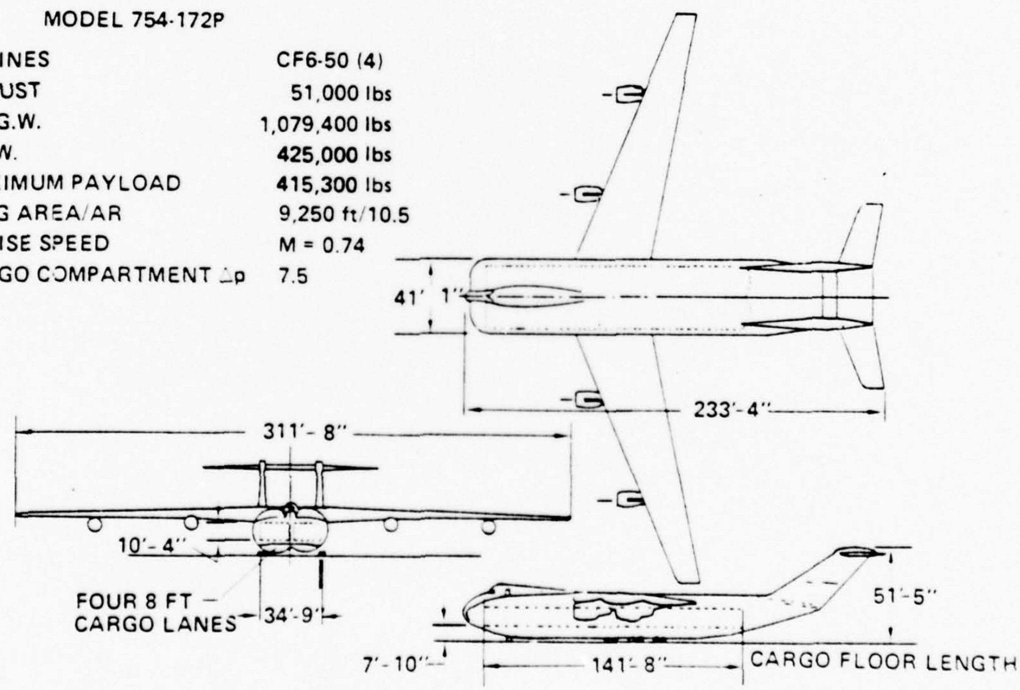


Figure 6-3. Typical Large Commercial Freighter

6.2.2 Performance Comparison

The impact of this weight savings was determined by calculating the increased payload capability, using MIL-C-5011A rules, and maintaining the design range at 6200 nmi. Figure 6-4 and 6-5 show the impact on payload range and fuel efficiency for the MIL-C-5011A and ATA mission rules, respectively.

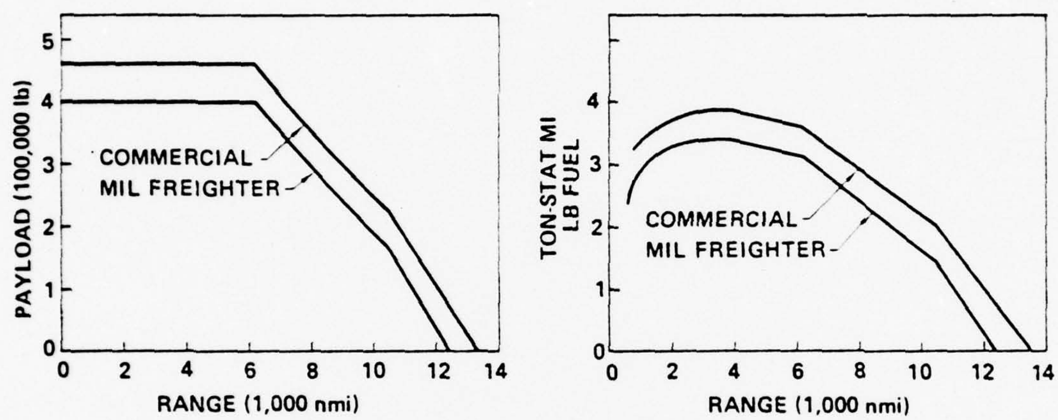


Figure 6-4. Commercial Derivative Comparison MIL-C-5011A Rules

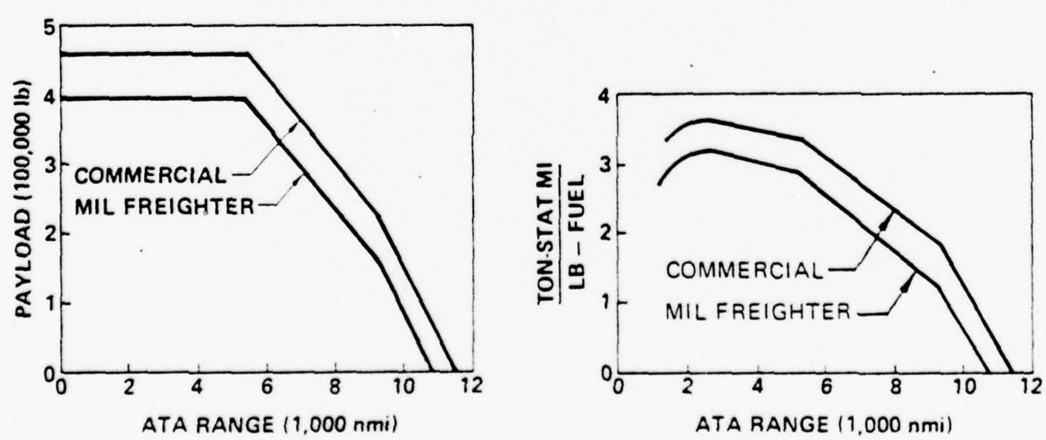


Figure 6-5. Commercial Derivative Comparison ATA International Rules

The military baseline, using MIL-C-5011A rules, the commercial derivative using ATA international rules, and the previously discussed commercial freighter are shown for comparative purposes on Figures 6-5 and 6-6.

It should be noted that the commercial freighter is designed for a different design point and has a technology base five years earlier. This technology base includes a derivative of existing engines; not a new engine as is the case of the 1985 technology postulated for the study baseline.

When comparing commercial capability, a commonly used figure of merit is Direct Operating Cost (DOC), which for freighters is expressed in terms of cents per ton mile (c/TM). The ton miles can be either available ton miles (ATM), which is based on a 100% load factor, or revenue ton miles (RTM), which assumes some realistic load factor. The yield is the price which the carrier charges the shipper and is composed of the DOC, the indirect operating costs (IOC), and carrier profit. Current yield on airfreight is running at a level of about 17 to 25 ¢/RTM depending on the type of equipment which the carrier operates. An advanced air freight system, which incorporates an improved containerized, inter modal container system, and which would greatly simplify ground handling and reduce costs, might have DOCs on the order of 5¢/RTM. Thus, reductions in DOC on the order of 0.1 ¢/RTM to 1.0 ¢/ATM are significant relative to the overall cost of shipping.

6.2.3 DOC Comparison

The baseline configuration was chosen on the basis of minimum DOC as has been explained previously.[1] The ATA formula contains provisions for showing the influence of most factors which influence the cost of acquiring and operating an air freighter.

Figure 6-6 summarizes the comparison of the Commercial Derivative operated with different rules and includes the commercial freighter for comparitive purposes.

The DOC was determined for the Baseline Commercial Derivative and is shown on Figure 6-7. The variation in DOC as payload is exchanged for range, as in Figure 6-6, is also shown. Of some interest is the insensitivity of DOC to operating at ranges less than the design points, as contrasted to operating at ranges greater than the design point. Also shown is a family of design points passing through the Commercial Derivative Baseline, illustrating the impact of design range on DOC. By designing to 6,200 nmi rather than to a more commercially desirable 2,750 nmi, the DOC was increased by about 0.5¢/ATM.

The baseline design criteria was for a cabin altitude of 18,000 ft. or 4.5 psi. The impact of imposing a 7.5 psi cabin pressure differential was examined, in order to examine the sensitivity of DOC to cabin altitude assumptions and further to provide a basis of comparison to the commercial freighter, which was designed for 7.5 psi. As shown on Figure 6-7, the decreased cabin altitude increased DOC by about 0.25¢/ATM or 5%.

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1 DOCs which were shown in the design sections were based on slightly different ground rules relative to the DOCs discussed in this section. These are more representative of anticipated values.

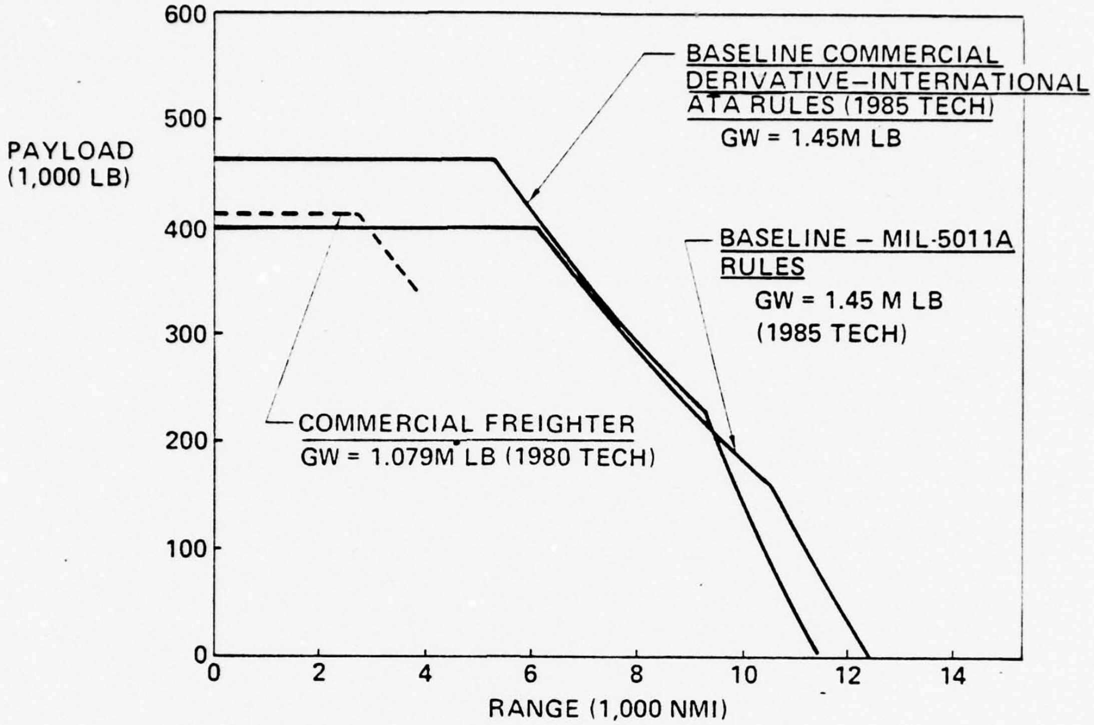

Figure 6-6 Baseline Commercial Derivative Performance

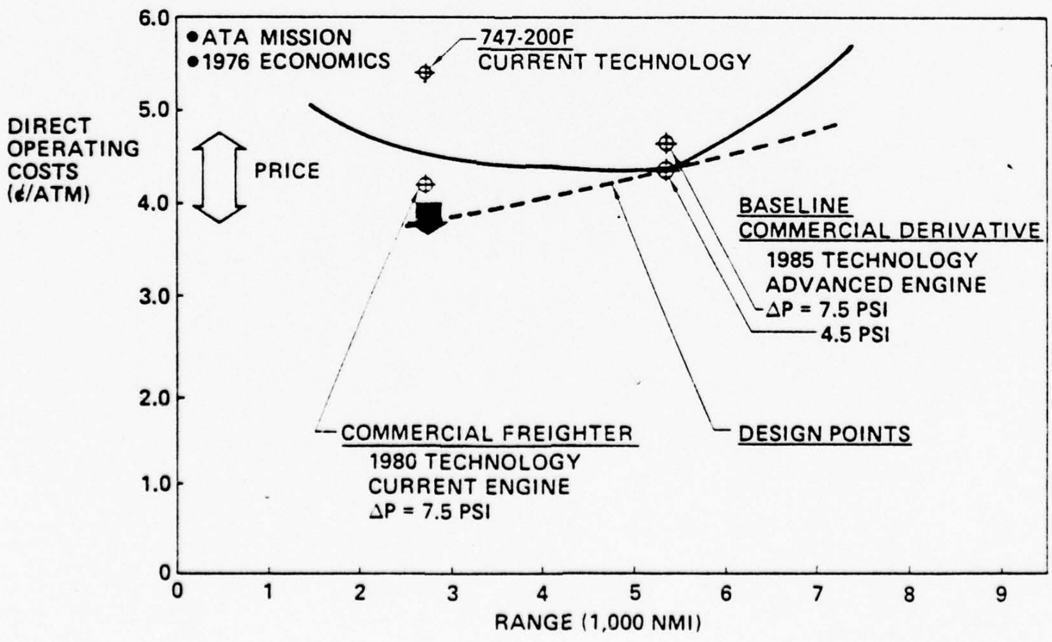

Figure 6-7 Commercial Commonality Direct Operating Costs

Also shown in Figure 6-7, for comparative purposes, are design points for the 747F and the commercial freighter mentioned earlier. Significant gains can be achieved by the technology advances included in the commercial freighter over that of current capability, reducing DOC by more than 20%. A technology level (1985) comparable to that of the IADS Baseline would further reduce the commercial freighter DOC to a level lower than the line described as Commercial Derivative Design points.

In summary, it appears that the projected technology base of 1985 produces significant gains in DOC over current technology and that a commercial derivative of a military transport is not incompatible with the needs for a commercial freighter. However, the long design range requirement imposed as a design criteria in this study imposes appreciable cost increase over a design range which is thought to be optimum from a commercial point of view.

7.0 System Cost and Evaluation

- Alternate fuels comparison

- Effect of fuel costs

- Detail baseline life cycle costs

- Technology and advanced design impact

7.0.0 SYSTEM COST AND EVALUATION

This Section contains a brief evaluation of the more significant aspects of the study. The cost effectiveness of alternate fuels, liquid hydrogen and liquid methane are compared to that of the JP Baseline and detailed life cycle costs are presented. The impact of fuel costs on design characteristics are also assessed. Particular emphasis is given to evaluating the effectiveness of advanced technology and design relative to the 1985 baseline and current technology.

7.1.0 SYSTEM COSTS

This section presents estimates of development, production, and twenty year operations and support costs for the JP, liquid hydrogen, and liquid methane fueled designs. Two different estimating techniques have been used to arrive at LCC costs for the three designs.

7.1.1 Approach

The life cycle costs in this analysis measure in FY 1976 dollars the costs of adding each of the three designs to the air lift inventory. Costs are peacetime only. Two sets of LCC are provided which are identified as Class 1 and Class 2. Rand Cost Models, Reference 15, were used to estimate the Class 1 airframe and engine costs. The Class 2[1] airframe and engine costs were estimated using Boeing cost models that are significantly more detailed than the Rand airframe model. Operations and support costs were estimated using the Air Force "CACE" model from AFR 173-10, Reference 16. Data on the C-141 provided the point of departure for the three IADS designs.

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[1] Class 2 costs are based on a detailed cost build-up based on structural components.

Included in the costing methodology are the costs of developing, producing, and operating each of the three designs. It is assumed that one developmental airplane for each design is procured, with the remainder of the flight test articles to be refurbished as production articles. A 250 airplane buy is assumed for each design, of which 225 are UE and 25 are command support. It is assumed that attrition would come out of the command support complement. Utilization rates are taken to be 1000 hrs/year.

7.1.2 <u>Ground Rules</u>

The IADS program plan is shown on Figure 7-1 with the ground rules on Table 7-1. Single source production is postulated due to the probable size of the program. Peak rate is 56 per year for the baseline design due to the physical size of the airplane. Development and production are assumed to be noncurrent due to Air Force review requirements. No cost penalty has been assessed to non-concurrence since it could be planned as nonconcurrent.

7.2.0 ALTERNATE FUELS COMPARISON

The configuration problem for designs which use liquid cryogenic fuels is generically different from those which utilize non-cryogenic fuels. Fuel tanks must not only be insulated to prevent excessive pressures, but must be an efficient pressure vessel. The double lobe body cross section developed in Reference 14, was selected because of natural geometric synergism between the body and the cryogenic fuel tank. The configurations for the LH_2 and LCH_4 designs are shown in Figures 7-2 and 7-3 respectively, with cryogenic tanks mounted on top of the double lobe fuselage.

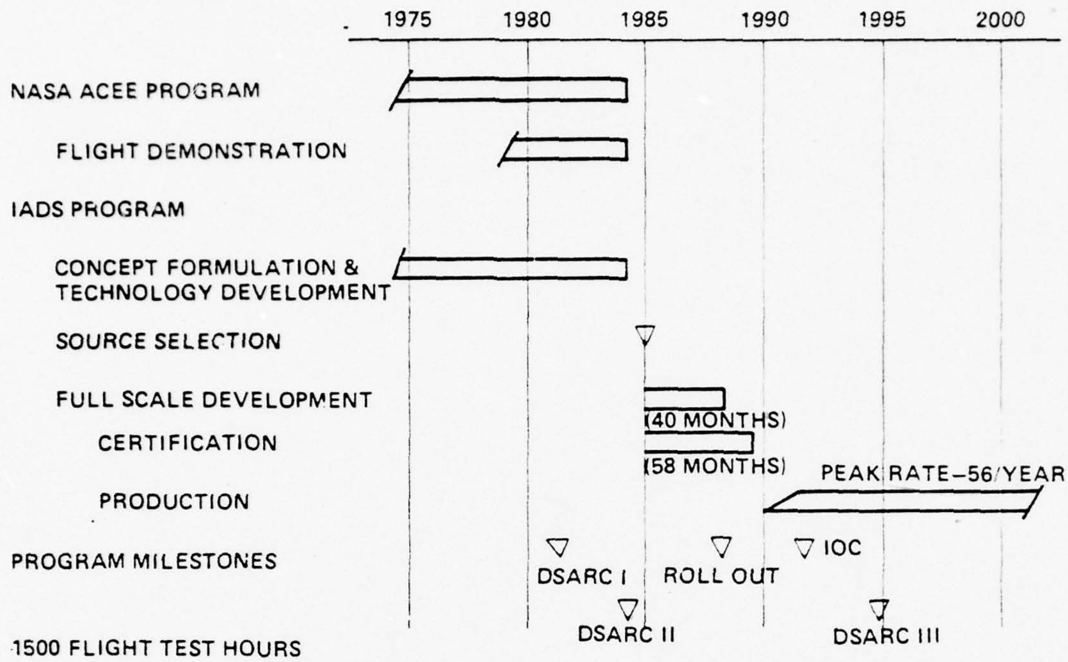

Figure 7-1. IADS Program Plan

Table 7-1. Life Cycle Cost Ground Rules

Model	Fuel	Rollout No. 1	Certification	Peak rate	Flight test hours
1044-013	JP	40 months	58 months	56/year	1,500
1044-015	LH_2	41 months	59 months	50/year	1,500
1044-016	LCH_4	42 months	60 months	42/year	1,500

RADIUS/EQUIV RANGE	3600/6200 nmi
TOGW	1,225,000 lb
OW	658,900 lb
PAYLOAD	400,000 lb
CRUISE SPEED M.80@	37,000 ft
WING AREA	9,007 ft
SWEEP, L.E.	26°
AR	15
ENGINES 4 @	67,700 SLST
TECHNOLOGY LEVEL	1985

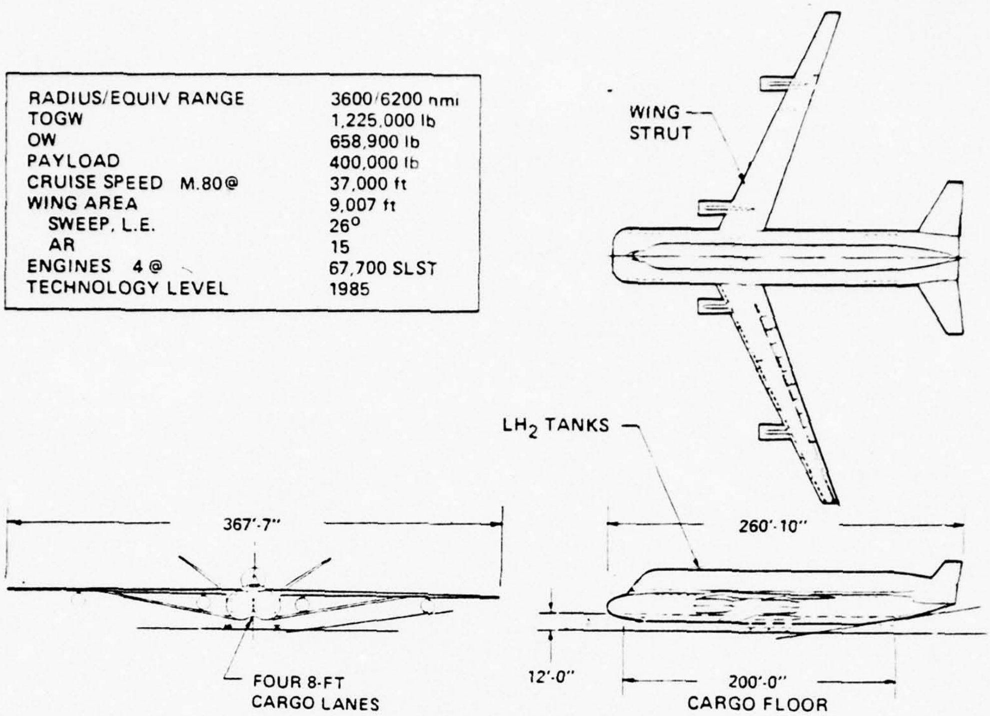

Figure 7-2. LH_2 Fueled Airplane

RADIUS/EQUIV RANGE	3600/6200 nmi
TOGW	1,800,000 lb
OW	908,450 lb
PAYLOAD	400,000 lb
CRUISE SPEED	M.80 @ 36,900 ft
WING AREA	13,235 ft
SWEEP, L.E.	26°
AR	15
ENGINES	6 @ 66,300 SLST
TECHNOLOGY LEVEL	1985

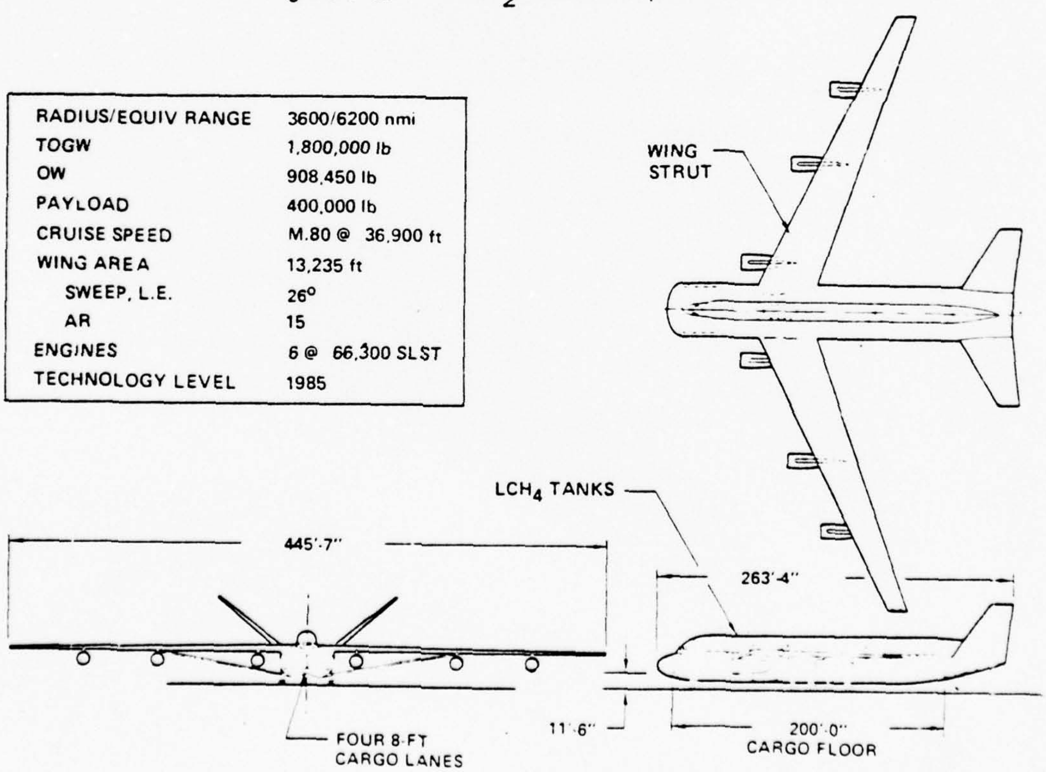

Figure 7-3. LCH_4 Fueled Airplane

Another characteristic of the cryogenic fueled designs is the absence of fuel from the wing caused by the internal pressure requirement. This produces an inefficient wing design because of the absence of the beneficial unloading effect of wing fuel during flight. As a result, and to show the cryogenic fuels in the best possible light, the wings were designed utilizing strut bracing to provide an advantageous structural arrangement. In other respects, the JP and cryogenic configurations are based on the same design practices and technology. The same engine cycle was utilized for the JP and cryogenic propulsion system, with appropriate changes being made to specific fuel consumption for their respective heating values. The heating values used were as specified below:

	Heating Value	Density
JP_4	18,400 BTU/lb	50.5 lb/ft^3
LH_2	51,570 BTU/lb	4.3 lb/ft^3
LCH_4	21,500 BTU/lb	26.5 lb/ft^3

Because of the high heating value and low density, the LH_2 design is significantly lighter than the other designs as shown in Figure 7-2. Because of lower density and a heating value only slightly higher than comparable to that of JP, the LCH_4 design was appreciably heavier than both the LH_2 and JP designs.

The comparison between the three designs utilizing JP, LH_2 and LCH_4 is made on the basis of LCC in Section 7.3.

7.2.1 Effect of Fuel Costs

The effect of varying design range and fuel costs on LCC and DOC for the JP fueled design is shown on Figure 7-4 and 7-5. Optimizing at the basic fuel

DESIGN POINT AIRPLANE DESCRIPTION

OPTIMIZED FOR	40 ¢ /GAL	80 ¢ /GAL
W/S	130 LB/FT	122 LB/FT
T/W	0.209	0.200
ΛLE	10°	10°
t/c	0.18	0.15
AR	8.9	9.6
BLOCK FUEL	505,000 LB	492,000 LB

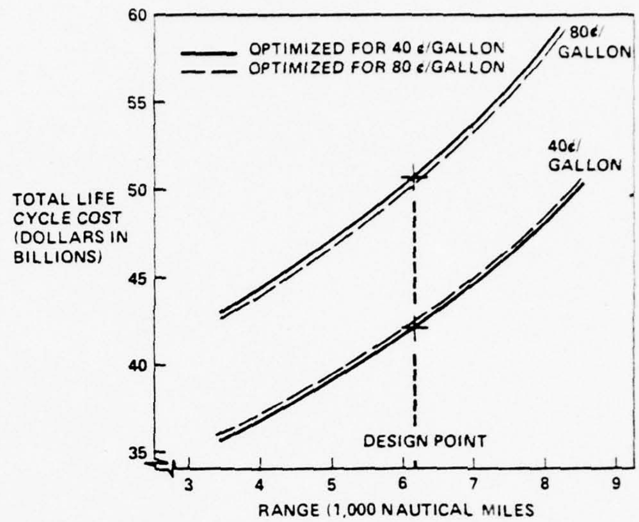

Figure 7-4. *System Cost: Effect of Fuel Cost on Life Cycle Cost*

DESIGN POINT AIRPLANE DESCRIPTION

OPTIMIZED FOR	40 ¢/GAL	80 ¢/GAL
W/S	136 LB/FT2	128 LB/FT2
T/W	0.221	0.214
ΛLE	26°	27°
t/c	0.10	0.10
AR	10.	10.6
BLOCK FUEL	460,000 LB	450,000 LB
LCC (40 ¢/GAL)	45.35	46.05
LCC (80 ¢/GAL)	54.05	54.40

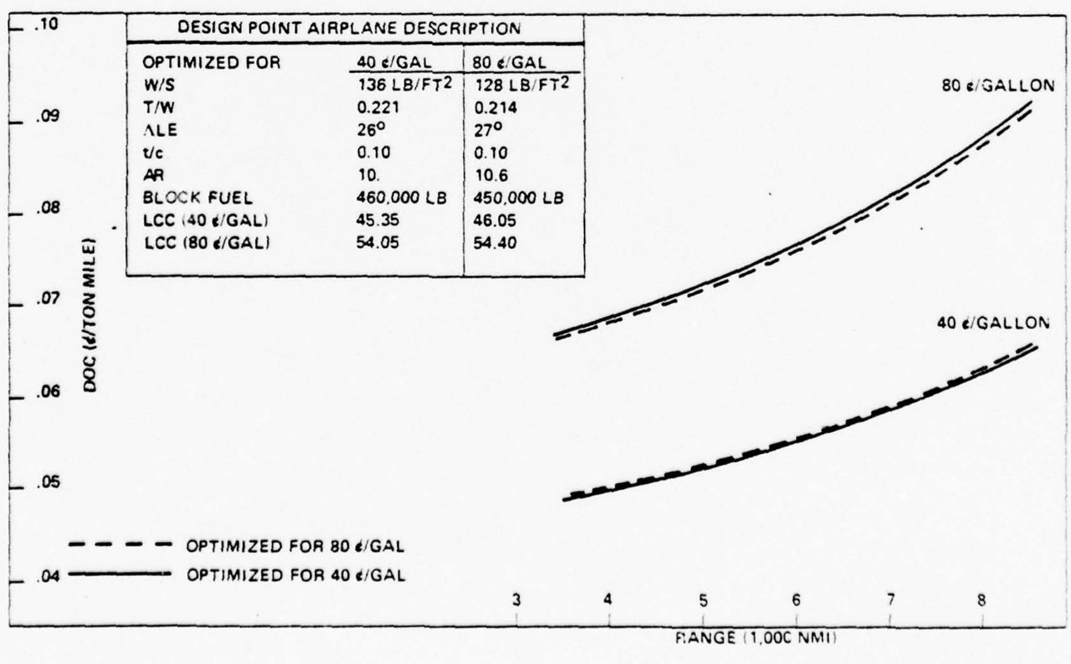

Figure 7-5. *System Cost: Effect of Fuel Cost on DOC*

price of 40c per gallon or at 80c per gallon does not have a significant effect on LCC. However, the fule cost increase itself would raise LCC $10.0 billon.

7.3.0 DETAILED BASELINE LIFE CYCLE COSTS

Tables 7-2, 7-3 and 7-4 provide detailed LCC data for the JP fueled, liquid hydrogen fueled, and liquid methane fueled designs, respectively. The Class 1 data is in Columns 1 and 2 and the Class 2 data is in Column 3. Class 1 data is differentiated between current technology and 1985 technology.

Table 7-2. Detailed Life Cycle Cost—JP

Cost element	Class 1 LCC current technology	Class 1 LCC 1985 technology	Class 2 LCC 1985 technology
Development			
Airframe	$1,562.0	$1,718.2	$3,007.1
Engines	446.5	491.1	834.2
Avionics	100.0	100.0	50.0
Flight test airplane			
Airframe	323.6	356.0	242.0
Engines	18.6	20.4	6.8
Avionics	2.0	2.0	2.0
Flight test operations	55.2	60.7	200.5
Total	$2,507.9	$2,748.4	$4,342.5
Production			
Airframe	$13,610.4	$14,971.5	$11,748.0
Engines	3,104.7	3,415.1	1,709.4
Avionics	500.0	500.0	500.0
Total	$17,215.1	$18,886.6	$13,957.4
Support investment			
Initial spares	$1,721.5	$1,888.7	$1,359.7
AGE, other	860.8	944.3	679.8
Total	$2,582.3	$2,833.0	$2,039.6
Operations and support			
AGE, spares, mods	$1,823.7	$1,823.7	$1,823.7
Military pay and allow	3,746.8	3,746.8	3,746.8
Depot maint	3,439.6	3,439.6	3,439.6
Fuel	9,453.7	9,453.7	9,453.7
Pipeline support	508.5	508.5	508.5
Other	1,997.4	1,997.4	1,997.4
Total	$20,969.7	$20,969.7	$20,969.7
Total life cycle cost	$43,275.0	$45,437.7	$41,309.2

Table 7-3. Detailed Life Cycle Cost—LH_2

Cost element	Class 1 LCC current technology	Class 1 LCC 1985 technology	Class 2 LCC 1985 technology
Development			
Airframe	$1,694.8	$1,864.2	$3,644.6
Engines	400.6	440.7	1,060.1
Avionics	100.0	100.0	50.0
Flight test airplane			
Airframe	356.2	391.8	286.0
Engines	15.9	17.5	5.0
Avionics	2.0	2.0	2.0
Flight test operations	60.2	66.2	227.4
Total	$2,629.7	$2,882.4	$5,275.2
Production			
Airframe	$14,965.2	$16,461.7	$13,354.0
Engines	2,655.8	2,921.3	1,265.5
Avionics	500.0	500.0	500.0
Total	$18,121.0	$19,883.0	$15,119.5
Support investment			
Initial spares			$1,511.9
AGE, other			755.9
Total	$2,718.1	$2,982.5	$2,267.9
Operations and support			
AGE, spares, mods	$1,897.0	$1,897.0	$1,897.0
Military pay and allow	3,880.2	3,880.2	3,880.2
Depot maintenance	3,039.0	3,039.0	3,039.0
Fuel	30,069.9	30,069.9	30,069.9
Pipeline support	522.0	522.0	522.0
Other	2,010.0	2,010.0	2,010.0
Total	$41,418.1	$41,418.1	$41,418.1
Total life cycle cost	$64,886.9	$67,166.0	$64,080.7

Table 7-4. Detailed Life Cycle Cost—LCH_4

Cost element	Class 1 LCC current technology	Class 1 LCC 1985 technology	Class 2 LCC 1985 technology
Development			
Airframe	$2,041.2	$2,245.3	$4,173.9
Engines	401.9	442.2	1,042.8
Avionics	100.0	100.0	50.0
Flight test airplane			
Airframe	449.8	494.7	331.1
Engines	22.7	24.9	9.4
Avionics	2.0	2.0	2.0
Flight test operations	74.0	81.4	264.4
Total	$3,091.6	$3,390.5	$5,873.7
Production			
Airframe	$18,802.5	$20,682.8	$16,388.9
Engines	3,788.1	4,166.9	2,372.3
Avionics	500.0	500.0	500.0
Total	$23,090.6	$25,349.7	$19,261.2
Support investment			
Initial spares	$2,309.07	$2,534.9	$1,926.1
AGE other	1,154.53	1,267.5	963.0
Total	$3,463.6	$3,802.4	2,889.1
Operations and support			
AGE, spares, mods	$2,299.0	$2,299.0	$2,299.0
Military pay and allow	4,101.0	4,101.0	4,101.0
Depot maintenance	3,917.7	3,917.7	3,917.7
Fuel	16,091.4	16,091.4	16,091.4
Pipeline support	544.5	544.5	544.5
Other	2,165.2	2,165.2	2,165.2
Total	$29,118.8	$29,118.8	$29,118.8
Total life cycle cost	$58,764.6	$61,661.4	$57,142.8

Development costs are non-recurring except for the flight test airplane airframe, engines and avionics costs which are recurring. Production airframe, engine and avionics costs are recurring. Support investment costs are also non-recurring.

Operations and support costs are held constant for each design across all three LCC estimates. Modification costs would probably vary as the production cost varies, however, the impact is not thought to be significant.

The Class 1 and Class 2 LCC estimates vary significantly within development and production. In the case of airframe costs this is due to the artificial method used to separate development from production costs for the Class 1 estimate. In the absence of separate development cost equations for engineering and tooling, the total cost versus airplane quantity curve at quantity 1 was taken as development cost.

Engine development and production costs vary significantly beween Class 1 and Class 2 methods. Here the Rand Model has separate equations for development and production as does the Class 2 method. The differences must be treated as a range of estimates with the higher values being most probable.

Avionics are treated the same for all estimates for all designs with the exception that development is $50 M lower for the Class 2 estimates. The actual development cost would probably fall in the range of $50 M to $100 M.

Support investment costs are assumed to be 10% of production cost for initial spares and 5% for AGE and other costs. This reflects current experience.

Operations and support costs are based on the C-141 data contained in AFR 173-10. In estimating the IADS designs' LCC the impact of increased size was one of the primary considerations. The other was particular fuel burn of the design being considered. The fuel costs provided in the work statement were used.

The results shown in Tables 7-2, 7-3, and 7-4 are shown graphically in Figure 7-6. The overwhelming factor of fuel costs, which are substantially higher for the cryogenic designs than for those of the JP design, indicates that based on the ground rules imposed by this study, cryogenics are substantially less cost effective than are JP based fuels. This conclusion can of course be challenged by the onset of future events which might well change the fuel costs in in favor of cryogenics.

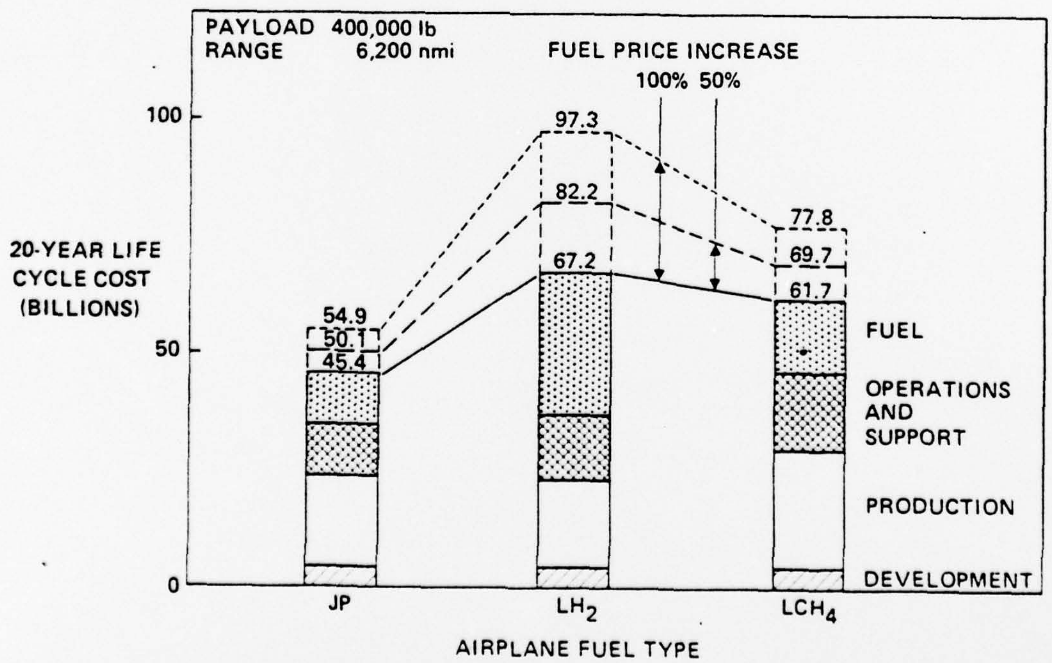

Figure 7-6. Total 20 Year Life Cycle Costs – Comparison of Alternate Fuels

7.4.0 TECHNOLOGY AND ADVANCED DESIGN IMPACT

The Innovative Aircraft Design Study (IADS) had as its objectives the identification of the most cost effective logistics aircraft configurations, and the identification of high leverage technologies. This subtask of Task II, Design Studies, was less concerned about evaluation than it was about design. However, some thought about the cost effectiveness of advanced design and technology was in order.

7.4.1 Gross Weight

In addition to considering the impact of advanced design and technology on the baseline, consideration was given to the effect some of the advanced technologies, such as composite primary structure, might have on the mission performance and effectiveness. Parametric studies had previously shown that for long range designs, high aspect ratio wings had definite benefits if they could be designed so as not to be too heavy. It was also shown that the take-off field length was generally providing the design criterion for thrust, and that the cruise thrust requirements were approximately 25% lower, making the use of a boosted takeoff attractive.

The combination of these design features, all of which appeared attractive, seemed to be a natural path to follow in the evaluation of advanced design and technology.

Figure 7-7 shows the design variation of range and gross weight for the baseline configuration, labeled minimum DOC. Also shown for comparison is the design to minimize fuel usage. Reductions in gross weight are shown which can be achieved by sequentially including in the design: (1) strut braced wings (AR = 15), (2) Composite wing and empennage, (3) boost engines. Reductions in gross weight of 200,000 to 300,000 pounds can be achieved by application of those advanced designs and technologies.

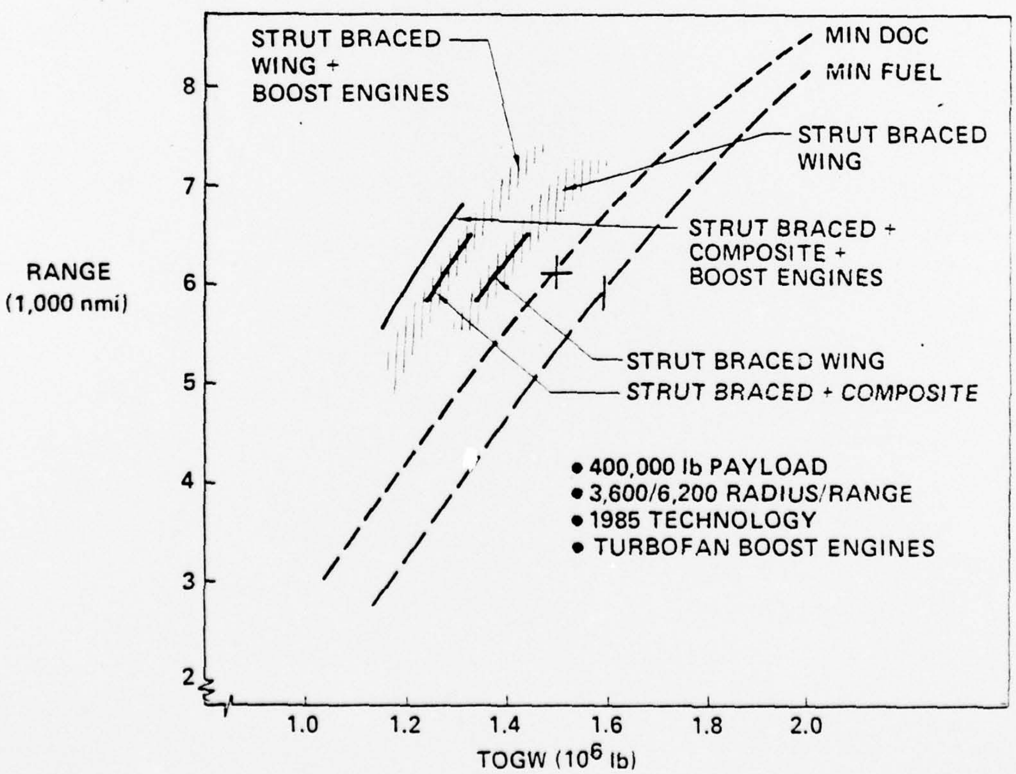

Figure 7-7 Innovative Designs: Baseline Configurations

7.4.2 Life Cycle Cost

These reductions in gross weight can be related to reductions in life cycle costs as shown in Figure 7-8. The improvement in cost effectiveness is shown by reductions in cost since the payload range and Mach number are held constant.

The current technology description is representative of current wide body designs with current generation of high bypass ratio engines. Various design improvements are added sequentially providing a group of low risk design improvements, including the baseline airplane. A reduction of more than 10 billion dollars might be achieved by a design incorporation of this sort over a new design which is based on current design practices. However, almost 3/4 of that increase in cost effectiveness would be achieved by the 1985 baseline alone.

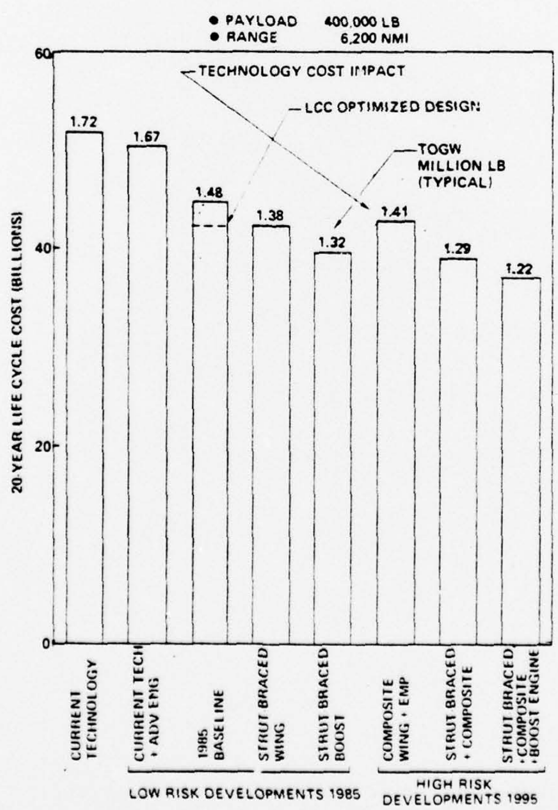

Figure 7-8. Advanced Design and Technology — Impact on Life Cycle Costs

The high risk developments, on the other hand, incorporated composite wing and empennage technology to achieve further benefits. However, increases in cost due to the increased complexity and general increased cost of technology must be considered which are included in the crosshatched areas.

Substantial uncertainty existed about how to evaluate the cost of technology. The cost of the composite technology was assessed to be 20% to 30% higher than that of the 1985 baseline. Although manufacturing labor was substantially reduced from that of conventional skin/stringer aluminum construction, the high cost of graphite composite material outweighed the labor reduction for a net increase. The net benefit due to the use of the composite appears to be marginal, based on the ground rules used in this study, which were: (1) 25% weight reduction in primary structure, (2) 15% reduction in secondary structure, (3) no composite in fuselage, and (4) $38/lb cost of composite material installed, based on $20/lb raw material.

However, life cycle cost, as has been pointed out, is a poor way to evaluate operational characteristics and performance of a military transport because of low utilization rates.

7.4.3 Fuel Efficiency

Better figures of merit than LCC may be fuel efficiency and DOC. Figure 7-9 shows the impact of the previously discussed design innovations on the ton miles/lb fuel relative to current wide body design and technology. As has been shown in other studies, laminar flow control has great potential as does, to a lesser degree, the prop fan. However, lower risk design innovation, when properly combined, may provide a factor of 2.0 in fuel efficiency over current designs.

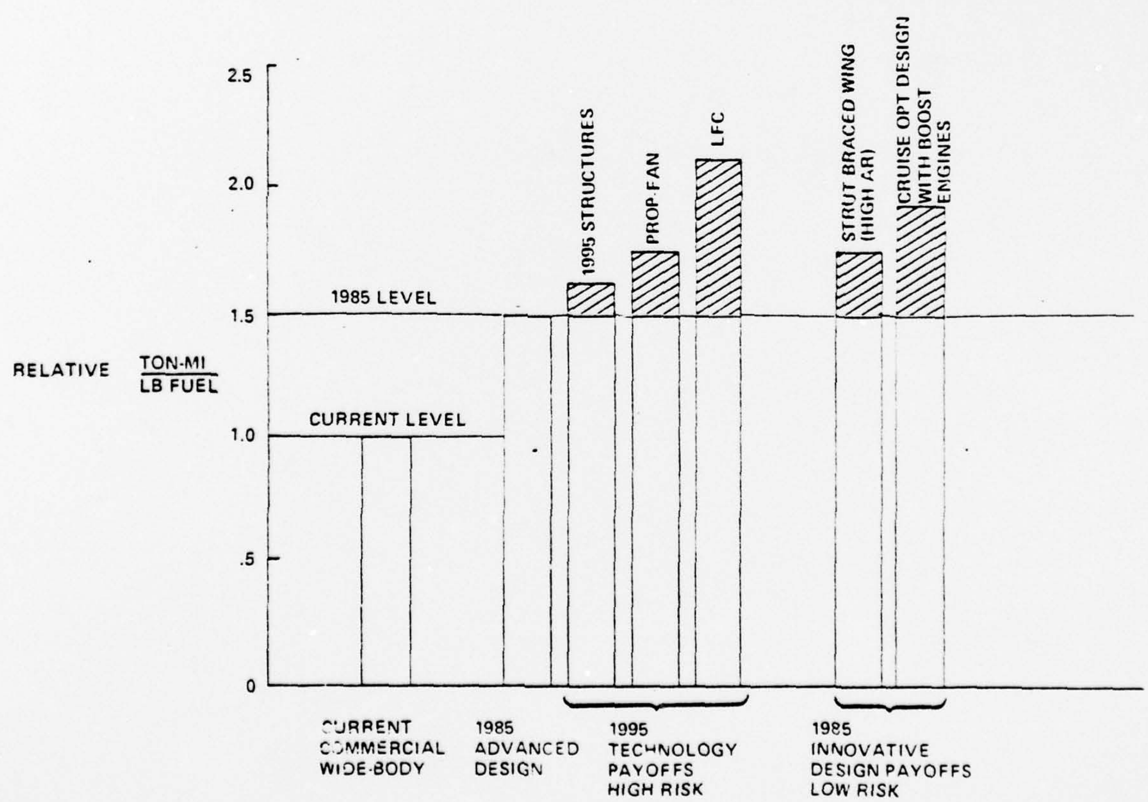

Figure 7-9. Advanced Design and Technology — Impact on Fuel Efficiency

In a similar vein, Figure 7-10 illustrates the impact of those previously discussed technologies on DOC. Reductions from the baseline commerical derivative of more than a .5c/ATM appear possible.[1]

7.4.4 DOC

Also shown in Figure 7-10 is the DOC of the Baseline Design, but with the circular cross-section body. It appears clear that if commercial commonality is a significant factor in the viability of a new transport program, serious consideration should be given to the compromises which must be made to the military requirement in favor of a more efficient fuselage structure as illustrated on Figure 7-10.

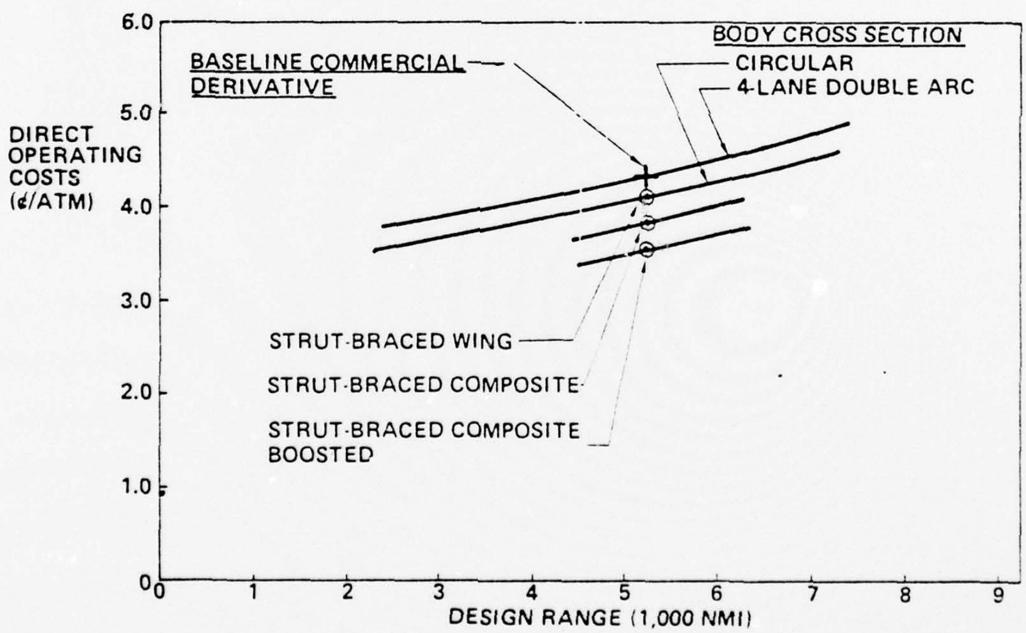

Figure 7-10. Advanced Design and Technology – Impact on DOC

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[1] These numbers are somewhat optimistic as they do not include the additional cost of composites.

8.0 Summary-
Conclusions and Recommendations

1. Advanced technology and design does pay
   - Fuel efficiency
   - Life cycle cost
   - DOC
2. No design problems were found as a result of validation
   - Flutter
   - Landing gear
3. Commercial derivative appears compatible
4. Cryogenics are not cost effective
   - Fuel cost

8.0.0 CONCLUSIONS AND RECOMMENDATIONS

The objective of this subtask of the IADS study was to examine the design problems associated with the definition of a new generation of strategic problems associated with the definition of a new generation of strategic transports and to identify those technologies which might have the most significant impact on system cost effectiveness. It was also desired to give consideration to the use of such a design to alternate missions such as Strategic Ballistic Missile Launch, Tactical and Command Control and Communication. Of particular interest was the application of the design to commercial use as a freighter.

8.1.0 ADVANCED TECHNOLOGY AND DESIGN

The 1985 technology base shows definite potential as a means of enhancing the cost effectiveness of a strategic transport. Most of the gains arise in small contributions from the various technologies but result in approximately 17% improvement in range factor and a 10% improvement in structural weight. These gains can produce a 50% increase in fuel efficiency and a 15% reduction in gross weight for airplanes of the size under consideration.

Advanced technology and design can increase those gains to 100% improvement in fuel efficiency and 30% decrease in gross weight.

A commercial derivative could expect DOC reductions from current wide body technology on the order of 30%.

8.2.0 DESIGN VALIDATION

The baseline selected was a conventional configuration and was selected to provide a well understood configuration upon which to extrapolate to high gross weight designs. A process of design validation which involved a more detailed examination of areas thought to be critical in nature: fuselage cross section, landing gear, and wing design, revealed no problems which would preclude consideration of a design which could perform the baseline mission of carrying a 400,000 pound payload on a 3,600 nmi radius mission.

8.3.0 COMMERCIAL DERIVATIVES

The configuration was selected to provide the same outsize capability as the C-5A and included such criteria as drive-through capability and minimum floor height. Those criteria have some penalties associated with them in terms of frontal area and fuselage weight. The design range of 6,200 mi, equivalent to the radius mission, imposes a significant penalty in terms of an increase in DOC of about 10% when compared to a nominal commercial design range of about 2,750 nmi.

Removal of equipment generic to the military mission which is easily removable amounts to about 10% in operational weight and might provide an attractive commercial derivative when coupled with the advantages of Government sponsored program.

8.4.0 Cryogenics Fuels

The fuel costs associated with cryogenic fuels drive the life cycle costs of the cryogenic designs. However, the future scarcity of JP types of fuels may make them attractive in the future.

8.5.0 Alternate Military Missions

Attractive capabilities exist in the use of a long range logistics transport for other missions. The large amounts of fuel, heavy payloads and large cargo volumes provide for great flexibility in the design of numerous military missions in addition to the logistics mission.

8.6.0 Recommendations

In order for the identified technologies to be developed in such a way that they are available for incorporation into an advanced military transport, a coherent development program is needed. This program should be integrated with the NASA ACEE program, but should provide the capability to integrate those technologies into a technology base which provides a solid foundation for the next military transport development effort.

Further work is needed to identify in greater detail the cost of the technology, including the manufacturing costs, in order to more clearly identify the cost effectiveness of such advanced technologies as advanced composite structure, prop fans and LFC.

The commercial market may well be more receptive to a design of smaller payload capability with a design range shorter than that selected for this study. Design studies should be instituted which examine the military and commercial attractiveness of a smaller class of designs focused on payloads of the 100,000 pound class.

The subject of commercial commonality is one of great complexity and requires additional effort. In particular the impact of assumptions concerning the modifications required to make a commercial derivative, and the significance of those assumptions to military system effectiveness is needed.

Recommendations

1. Integrated NASA/USAF technology program
   - Technology demonstration
   - Technology costs
2. Consider smaller designs
   - Commercial market
3. Continue indepth examination of commercial commonality

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